NATIONAL TRANSPORTATION SAFETY BOARD
Investigative Hearing
US Army PAT25, Sikorsky UH-60L and
PSA Airlines flight 5342
MHI RJ Aviation CL-600-2C10 (CRJ700), N709PS
Midair Collision in Washington, DC
January 29, 2025

Docket No.	**SA-544**
EXHIBIT	
7-HELO-A	

Helicopter Airworthiness Group Chairman's Factual Report

(74 Pages)

National Transportation Safety Board
Office of Aviation Safety
Washington, DC 20594

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HELICOPTER AIRWORTHINESS

Group Chair's Factual Report

June 2, 2025

A ACCIDENT

Location:	Washington, District of Columbia
Date:	January 29, 2025
Time:	2048 eastern standard time
Helicopter:	Sikorsky UH-60L, US Army tail number 00-26860
Airplane:	MHI RJ Aviation CL-600-2C10, registration N709PS

B HELICOPTER AIRWORTHINESS GROUP

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C SUMMARY

On January 29, 2025, about 20:48 eastern standard time (EST), a Sikorsky UH-60-L helicopter, operated by the United States Army under the callsign PAT25, and an MHI (Mitsubishi Heavy Industries) RJ Aviation (formerly Bombardier) CL-600-2C10 (CRJ700) airplane, N709PS, operated by PSA Airlines as flight 5342, collided in flight approximately 0.5 miles southeast of Ronald Reagan Washington National Airport (DCA), Arlington, Virginia (VA), and impacted the Potomac River in southwest Washington, District of Columbia (DC).

The 2 pilots, 2 flight attendants, and 60 passengers aboard the airplane and all 3 crew members aboard the helicopter were fatally injured. Both aircraft were destroyed as a result of the accident. Flight 5342 was operating under the provisions of Title 14 *Code of Federal Regulations (CFR)* Part 121 as a scheduled domestic passenger flight from Wichita Dwight D. Eisenhower National Airport (ICT), Wichita, Kansas, to DCA. PAT25 originated from Davison Army Airfield (DAA), Fort Belvoir, VA, for the purpose of the pilot's annual standardization evaluation with the use of night vision goggles (NVGs). Night visual meteorological conditions prevailed in the area of DCA at the time of the accident.

D DETAILS OF THE INVESTIGATION

1.0 Helicopter Information

1.1 Helicopter Description

The Sikorsky UH-60L helicopter is a military helicopter that does not have a Federal Aviation Administration (FAA) standard airworthiness certificate.[1] The UH-60L has a four-bladed, fully-articulated main rotor system that provides helicopter lift and thrust, and a four-bladed, fully-articulated tail rotor that provides directional control of the helicopter. The tail rotor is canted such that tail rotor thrust also provides lift for the aft section of the helicopter. The UH-60L has a nonretractable wheel landing gear in a reverse tricycle configuration. The UH-60L is equipped with two GE Aerospace T701-series turboshaft engines.

In this report, all left, right, up, and down orientations as well as clock positions are in the aft-looking-forward frame of reference unless otherwise specified. In this report, the left seat pilot is "left pilot" and right seat pilot is "right pilot". In some of the technical documentation provided by manufacturers, the left seat pilot is identified as the "copilot" and the right seat pilot is identified as "pilot". Other NTSB reports for this investigation may identify the left seat pilot as "pilot" and the right seat pilot as "instructor pilot".

1.2 Accident Helicopter History

The accident helicopter was manufactured by Sikorsky and delivered to the US Army in March 2001. The accident helicopter was Sikorsky airframe serial number (S/N) 702614 and US Army tail no. 00-26860. At the time of the accident, tail no. 00-26860 was assigned to the 12th Aviation Battalion (12th BN), located at DAA.

2.0 On Scene and Post-Recovery Wreckage Examination

On January 30, 2025, members of the Helicopter Airworthiness Group convened at DCA to begin organizing investigative activities and examined pieces of helicopter wreckage that was recovered overnight. On January 31, 2025, the combination cockpit voice recorder and flight data recorder were found, subsequently recovered, and transported to the NTSB Recorders Lab in Washington, DC. From February 1 to 6, 2025, the Helicopter Airworthiness Group reviewed the aircraft logbook and modifications accomplished on the helicopter in addition to examining pieces of helicopter wreckage as they were recovered. On February 6, 2025, the helicopter main wreckage was recovered from the Potomac River. On

[1] As of the date of this report, the FAA Dynamic Regulation System (DRS) contained the following numbers of restricted category type certificates for military surplus H-60 helicopters: 4 for EH-60A; 18 for UH-60A; 2 for HH-60L; and 4 for UH-60L.

February 7, 2025, the recovered helicopter main wreckage was transported to Hangar 7 at DCA. From February 8 to 11, 2025, members of the Helicopter Airworthiness Group examined the helicopter main wreckage as well as additional pieces of helicopter wreckage that were recovered.

2.1 On Scene Observations

The accident helicopter main wreckage was located about 38° 50′ 27.39″ N by 77° 1′ 32.04″ W (**figure 1**). The helicopter came to rest inverted with its right side lower than its left side (**figure 2**). The tail structure had partially separated from the fuselage near the tail transition section. During low tide, the left wheel, tail wheel, tail boom, left horizontal stabilizer, and belly of the helicopter were visible. During high tide, the majority of the helicopter was submerged.

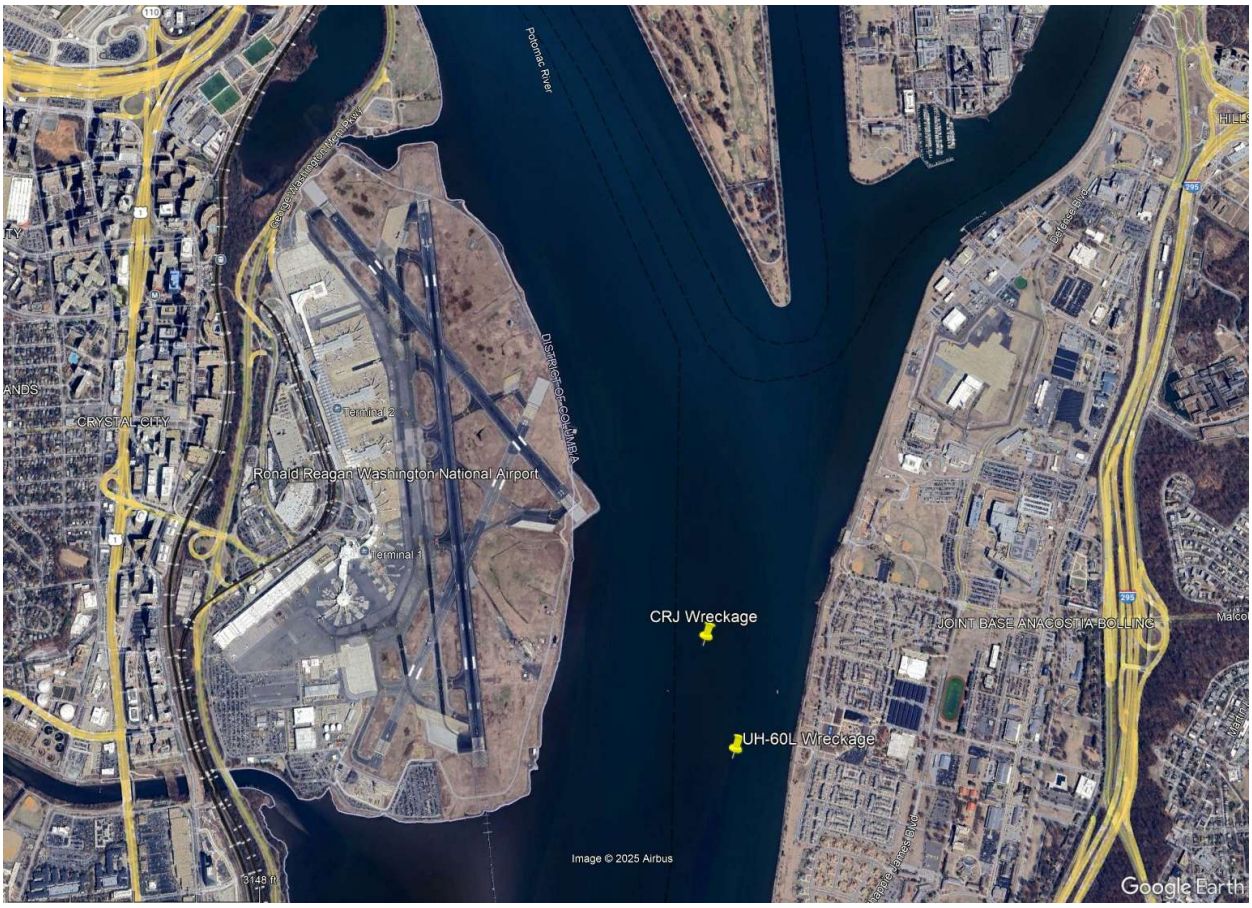

Figure 1. The location of the accident helicopter and airplane wreckage.

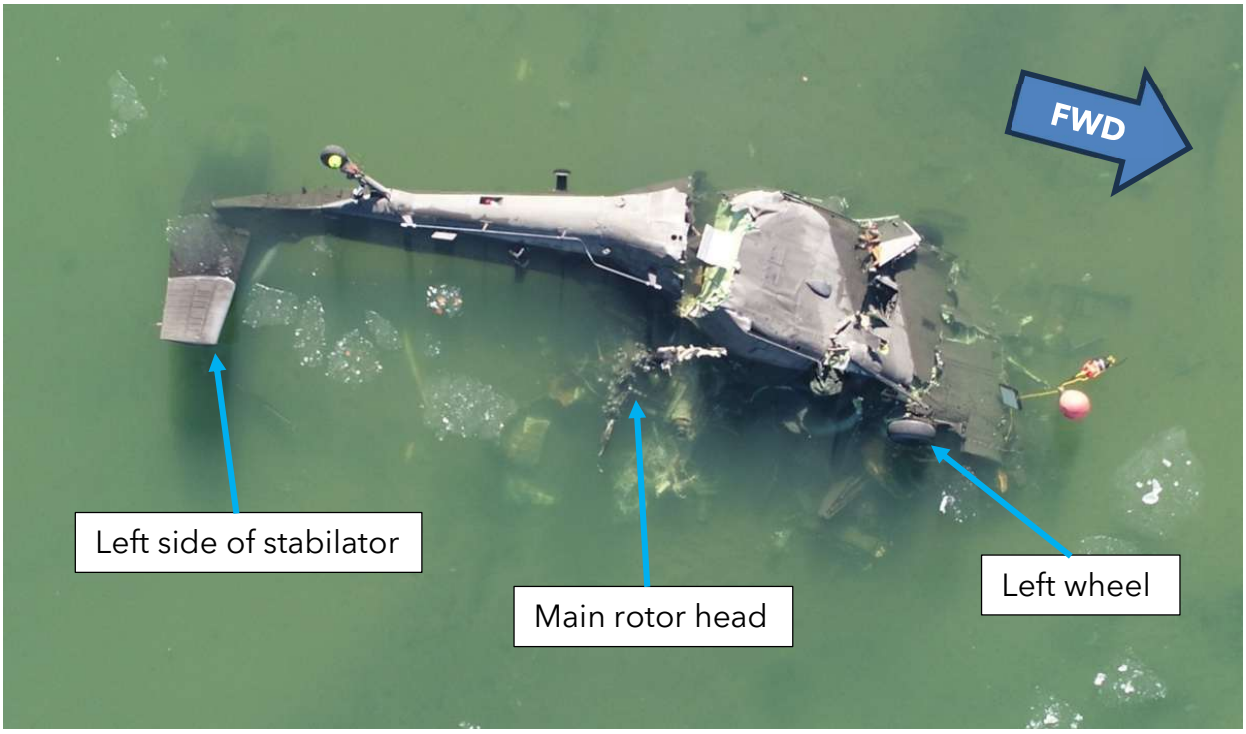

Figure 2. The helicopter main wreckage in the Potomac River.

2.2 Structures

2.2.1 System Description

The airframe of the UH-60L helicopter is divided into six sections: cockpit, cabin, transition, tail cone (tail boom), tail rotor pylon (vertical fin), and main rotor pylon (**figure 3**). In this report, the "main fuselage" is composed of the cockpit, cabin, main rotor pylon, and transition. The primary structure is constructed of aluminum. Titanium and steel are used for firewalls and various fittings. Nonstructural members are primarily constructed of reinforced plastic. Two internal crash-resistant fuel tanks (CFT) are located in the transition section.

The non-retractable landing gear system is a reverse-tricycle configuration, with the two main landing gear (MLG) struts attached to the cockpit-cabin area and a tail wheel strut connected to the tail cone and tail rotor pylon.

The external stores support system (ESSS) are two wing-like structures that are installed on the upper cabin frame of the helicopter and are each supported by two struts that attached to the lower cabin frame. Each ESSS wing has two mounting sponsons, an inboard and an outboard sponson. Items, such as external CFTs or weapons, can be attached to each sponson. The accident helicopter had the ESSS installed at the time of the accident, with an external CFT attached to each outboard mounting sponson.

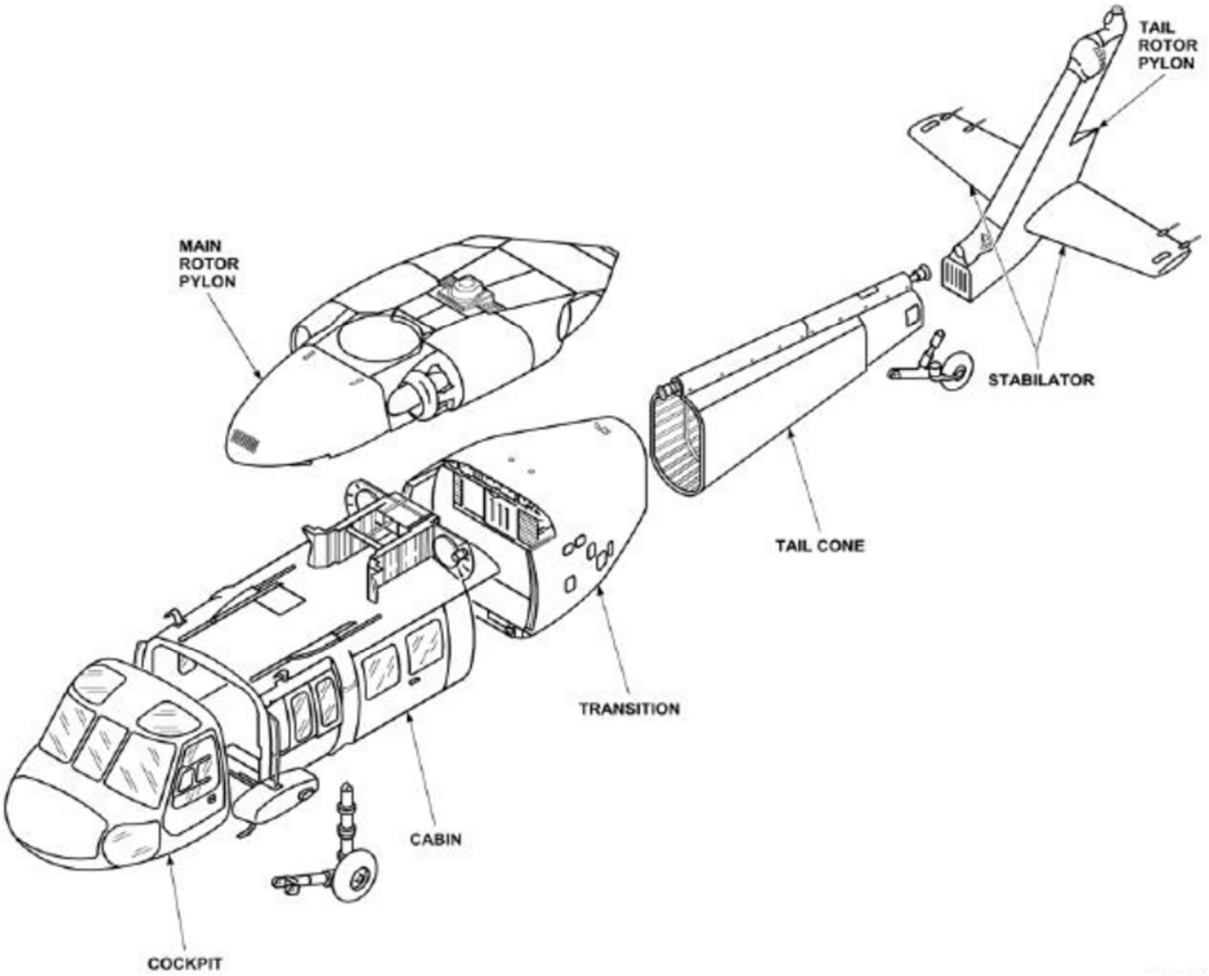

Figure 3. Diagram of the airframe sections. (Image courtesy of Sikorsky)

2.2.2 Structures Wreckage Observations

The main fuselage structure had multiple fractures and was primarily held together during recovery by cabling, hoses, and remnant structure (**figure 4**). The cockpit area was heavily fragmented. The roof structural beams, to which the transmission and primary servos attach, had separated from the main fuselage. The left and right vertical structural beams between frame station (STA) 295 and 308, between the gunner's window and the cabin sliding door, had also fractured and separated from the main fuselage and the top and bottom ends. The cabin floor fractured in multiple locations. The cargo hook was found separated from the recovered main wreckage. The two internal CFTs were present in their normally installed location. The right internal CFT was partially detached from the airframe. The left internal CFT remained attached to the airframe.

Figure 4. The helicopter wreckage recovery from the Potomac River.

Various pieces of composite structure from the airframe and engine cowlings, as well as pieces of the engine firewalls, were recovered separated from the main wreckage. The sliding cowling, normally located at the forward end of the main rotor pylon, was found separated in the recovered wreckage and exhibited multiple fractures, primarily at its front end. The upper wire strike protection system cutter remained installed on the forward-top sliding cowling. The left and right engine cowlings, upper cowlings (in the vicinity of the oil cooler behind the main transmission, had separated into multiple pieces.

The left cockpit door had separated from the main wreckage and had fractured at its forward-upper corner. Two cabin sliding doors were recovered separated from the main wreckage. One cabin sliding door was whole and did not exhibit significant fragmentation or deformation. The second cabin sliding door had fractured into several large pieces and its forward and upper portions exhibited significant impact deformation.

The transition section was fractured and partially separated at its forward side, but mostly intact near its aft side. The tail boom was separated from the main fuselage during the wreckage recovery process. The tail boom was whole and exhibited damage limited primarily to wrinkled skin. The vertical fin was mostly intact except for its upper end, near where the tail rotor is located, which had fractured and

separated from the rest of the vertical fin. The upper end of the vertical fin was deformed to the [helicopter] right direction (**figure 5**). The stabilator was whole and remained installed on the vertical fin. The stabilator exhibited an impact deformation in the forward direction located about the middle and aft end of the stabilator. The stabilator position was near 0 degrees.

Figure 5. The vertical fin showing the top portion that was deformed to the right. This image is in the forward-looking-aft frame of reference.

The search light (underneath right pilot's seat) and landing light (underneath left pilot's seat) were present in the cockpit wreckage but were crushed and deformed from impact. The left ESSS light remained installed on the outboard mounting sponson. The left ESSS light glass cover and bulb were fractured. The right ESSS light mount had separated from the right ESSS outboard mounting sponson. The right ESSS light glass cover and bulb were fractured. The lower anticollision light remained installed and was intact. The tail position light remained installed on the top of the vertical fin piece, but the light was fractured from impact. The tail anti-collision light was not found in the vertical fin wreckage.

The left and right MLG were present near their normally installed locations and attached to their airframe attachment points, but the structure which the landing gear attaches to had fractured and partially separated from the airframe. The tail wheel remained installed on the tail boom. The struts and pistons for the left and right MLGs and tail wheel did not exhibit significant deformation. The left MLG tire was measured

about 114 psi. The right MLG tire was measured at 105 psi. The tail wheel tire was measured at 100 psi.

The left and right external CFTs had separated from the ESSS and the aft fins had separated from the external CFTs. The left ESSS was attached to the main fuselage wreckage with the inboard mounting sponson still attached to the ESSS structure while the outboard mounting sponson was only attached via hoses and cables. The right ESSS was partially attached to the recovered wreckage with the inboard mounting sponson still attached to the ESSS structure. The outboard mounting sponson had completely separated from the right ESSS and was recovered separate from the main fuselage wreckage.

2.3 Main Rotor System

2.3.1 System Description

The UH-60L helicopter has a four-bladed, fully-articulated main rotor system. The main rotor hub is constructed of titanium. Each main rotor blade attaches to the hub via a spindle assembly that retains the blade to the hub and contains elastomeric spherical and thrust bearings to accommodate blade pitch, flap, and lead-lag movements. The main rotor head oleo-dampers, one for each blade, provide dampening for blade lead-lag movements. A pitch horn and an oleo-damper connect to each blade spindle assembly. The main rotor blades are constructed of a titanium spar and a composite afterbody[2] composed of a fiberglass skin, Nomex honeycomb core, and a graphite and fiberglass trailing edge. The main rotor blades have a titanium abrasion strip at its leading edge, with the outboard portion of the blade having a nickel abrasion strip. The main rotor blades are identified by color, presented in the order of rotation, as 'red', 'blue', 'yellow', and 'black'. Droop stops prevent the main rotor blades from excessively drooping. A bifilar assembly is installed on the main rotor hub and to provide rotor vibration damping.

Power from each engine is transmitted to their respective input module, the latter of which is part of the main transmission system, through a high speed shaft (HSS). A freewheeling unit, installed at the input side of each input module, allows for each engine to disconnect from the rotor drive train when the engine power turbine speed (Np) is below main rotor speed (Nr). Each input module provides power to an accessory module, installed at the front end of the input module, that powers a hydraulic pump and an electrical generator. Each input module transmits power to the main gearbox (MGB). The MGB provides drive to the main rotor shaft and the tail takeoff drive pinion, the latter of which powers the tail rotor drive train.

[2] The afterbody is the blade structure behind the blade spar.

2.3.2 Main Rotor Drive System Wreckage Observations

The MGB remained attached to the structural roof beams, but the structural roof beams had separated from the main fuselage (**figure 6**). The MGB housing was whole and the main rotor shaft remained installed. The left input module housing remained attached to the MGB, but its housing had fractured, exposing its output bevel gear and accessory drive gear. The left input module's input pinion was not attached to the input module and was not found. The left HSS was not found. The left gimbal was found attached to a fractured portion of the forward support tube, both of which were found separate from the recovered main wreckage. The left gimbal's forward end was fractured and deformed. The left accessory module was found separated from the main wreckage. The left generator and left hydraulic pump remained attached to the left accessory module.

Figure 6. The MGB, main rotor shaft, and main rotor head. The structural roof beams are underneath the MGB.

The right input module had separated from the MGB at its flanged attachment. The right input module and the right accessory module were found separated from the main wreckage. The right generator remained attached to the right accessory module, but the portion of the accessory module with the hydraulic pump had

fractured and separated from the accessory module. The right gimbal, normally between the input module and the forward support tube, had separated from the forward support tube and a remnant portion remained installed on the right input module. The right support tube remained installed with the recovered right engine, and the right HSS remained installed on the right engine.

2.3.3 Main Rotor Wreckage Observations

The main rotor hub remained installed on the main rotor shaft. The bifilar remained attached on the main rotor hub. One bifilar arm and its weight remained attached to the bifilar hub, but the remainder of the bifilar arms had fractured and separated from the bifilar. One separated bifilar arm, with its weight still attached, was recovered separated from the main wreckage. The other two bifilar arms and their weights were not found. The main rotor hub cap, which contains the blade de-icing connectors, was found separated from the main rotor hub. One of the four de-icing connectors remained installed on the cap.

An inboard portion of the ‘red’ main rotor blade assembly containing the blade cuff and root end of the blade spar; an outboard portion of the spindle assembly that had fractured from its parent spindle assembly; the pitch horn; and a portion of the ‘red’ pitch control rod (PCR) was recovered separated from the main rotor wreckage. The ‘red’ spindle assembly had fractured immediately adjacent to the outboard face of the elastomeric spherical bearing. The ‘red’ PCR upper rod end remained attached to the pitch horn and the link body was continuous for about 15.5 inches, where it was fractured at its lower end. All observed fractures showed signatures of overload. The ‘red’ damper remained attached to the main rotor hub but had fractured at the piston’s threaded end for the rod end that connects to the ‘red’ spindle assembly. The spindle-side rod end remained installed on the separated ‘red’ spindle assembly. The ‘red’ droop stop had fractured from the hub-side spindle assembly, but the droop stop had fractured and separated. The inboard end of the ‘red’ main rotor blade showed S/N C302-18630. According to Sikorsky main rotor blade production records, S/N C302-18630 is associated with main rotor blade S/N A007-16549.

The ‘blue’ spindle remained attached to the main rotor hub. The ‘blue’ blade cuff and remnants of the ‘blue’ blade remained attached to the spindle. The ‘blue’ damper remained installed on the main rotor hub but had fractured at the piston’s threaded end for the rod end that connects to the ‘blue’ spindle assembly. The spindle-side rod end upper lug had fractured in overload and the rod end was not present. The blade inspection method (BIM) indicator for the ‘blue’ blade remained installed and showed white. The ‘blue’ blade flap and droop stops remained installed on its spindle assembly. The springs for the ‘blue’ droop stop had separated from the stop. The inboard end of the ‘blue’ main rotor blade showed S/N C302-21428. The blue main rotor blade was S/N A007-18598.

The ‘yellow’ spindle remained attached to the main rotor hub. The ‘yellow’ blade cuff and remnants of the ‘yellow’ blade remained attached to the spindle. The ‘yellow’ damper remained installed on the main rotor hub but had fractured at the piston’s threaded end for the rod end that connects to the ‘yellow’ spindle assembly. The spindle-side rod end remained installed on the ‘yellow’ spindle assembly. The BIM indicator for the ‘yellow’ blade had separated and was not recovered. The ‘yellow’ blade flap and droop stops remained installed on the spindle assembly. The springs for the ‘yellow’ droop stop had separated from the stop. The inboard end of the ‘yellow’ main rotor blade showed S/N B302-03041. According to Sikorsky main rotor blade production records, S/N B302-03041 is associated with main rotor blade S/N A007-03686.

The ‘black’ spindle remained attached to the main rotor hub. The ‘black’ blade cuff and remnants of the ‘black’ blade remained attached to the spindle. The ‘black’ damper remained installed on the main rotor hub but had fractured at the piston’s threaded link for the rod end that connects to the ‘black’ spindle assembly. The spindle-side rod end remained installed on the ‘black’ spindle assembly. The BIM indicator for the ‘black’ blade remained installed and showed white. The ‘black’ blade flap and droop stops remained installed on its spindle assembly. The advancing-side spring for the ‘black’ droop stop remained installed but the trailing-side spring was disconnected from the stop. The inboard end of the ‘black’ main rotor blade showed S/N C302-12200. The ‘black’ blade was S/N A007-13223.

A piece of main rotor blade spar, about 50 inches in length, was found separate from the main wreckage and exhibited a straight appearance except for its fractured ends, which were deformed in the chordwise direction. This 50-inch long piece had remnant skin attached to it but was missing its leading edge abrasion strip. A piece of main rotor blade spar, about 5 feet in length, was found separate from the main wreckage and was partially fractured and bent in the chordwise direction. These spar pieces were missing their leading edge tungsten weights. Various pieces of main rotor blade afterbody was found within the recovered wreckage. A section of main rotor blade, about 17.5 feet in length, was also recovered. This 17.5-foot long section of main rotor blade contained the spar and a portion of the afterbody. The bulk of the tip end was present in this 17.5-foot long section and the tip cap showed S/N B849-16626. According to the aircraft logbook, based on the tip cap S/N, it was installed on main rotor blade S/N A007-16549, which was consistent with the ‘red’ main rotor blade.

2.4 Tail Rotor System

2.4.1 System Description

The tail rotor drive system is powered by the MGB tail takeoff pinion (**figure 7**). Power is ultimately transmitted to the tail rotor via six tail rotor drive shafts (TRDS), the

axial fan drive shaft, and two gearboxes. Each TRDS is composed of a shaft tube and a three-lobed flange that is riveted at each end of the shaft tube. The No. 1 TRDS[3] is connected at its front end to the tail takeoff pinion and is connected to the axial [oil cooler blower] fan at its aft end. The axial fan shaft is connected to the axial fan at its forward end and to the No. 2 TRDS at its aft end. The Nos. 2 through 5 TRDS[4] are connected in succession between the axial fan and the intermediate gearbox (IGB), the latter of which is located at the base of where the tail boom meets the vertical fin. Viscous dampers provide damping and support between the Nos. 2 through 5 TRDS The No. 6 TRDS[5] is connected between the IGB and tail gearbox (TGB). Laminated flexible couplings, located between connection points along the tail rotor drivetrain, transit drive between TRDS and accommodate minor axial and angular misalignment.

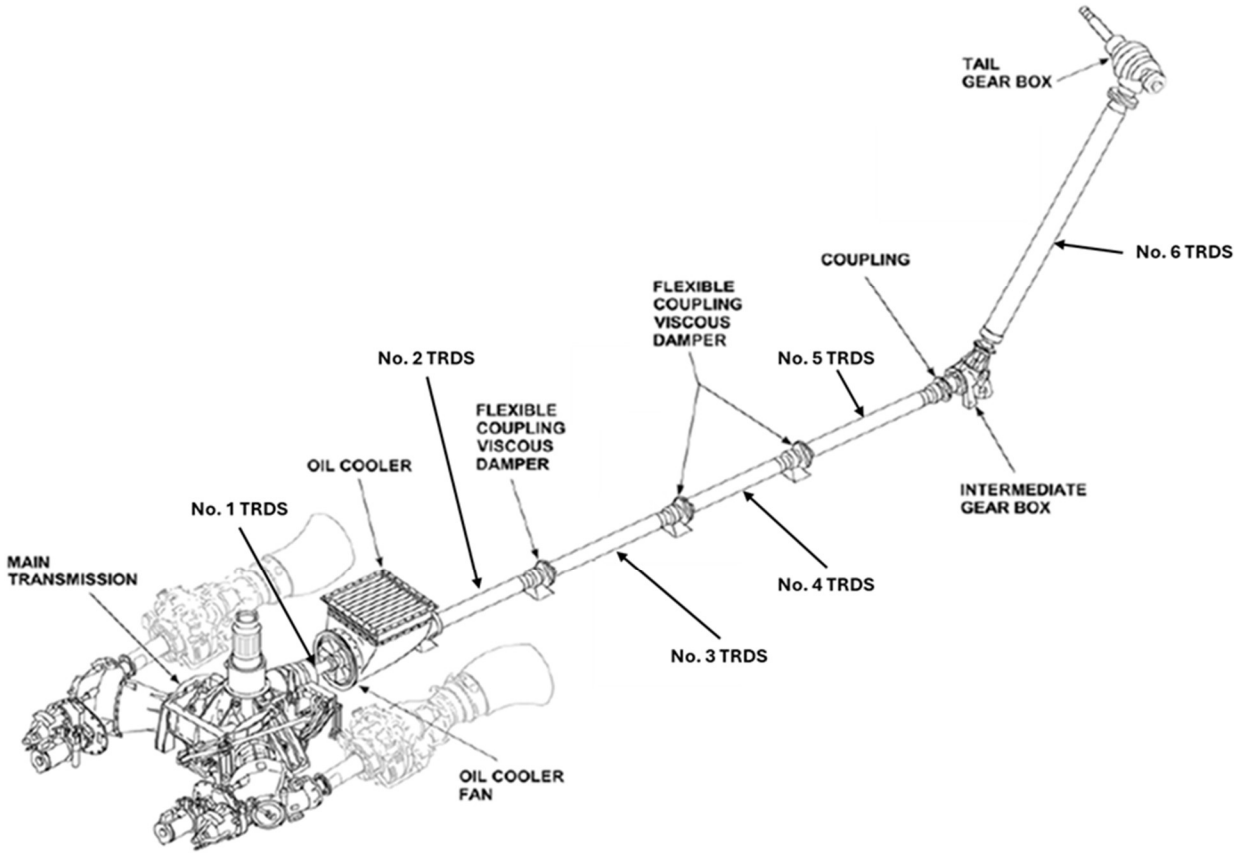

Figure 7. Diagram of the transmission and drive system components. (Image courtesy of Sikorsky)

The tail rotor is composed of two paddle assemblies that are set 90 degrees apart. The graphite paddle acts as the blade spar and can twist, via elastomeric bearing supports, to accommodate blade pitch changes. The two tail rotor paddles

[3] The No. 1 TRDS is also known as the Section I drive shaft.
[4] The Nos. 2 through 4 TRDS assembly is also known as the Section II drive shaft. The No. 5 TRDS is also known as the Section III drive shaft.
[5] The No. 6 TRDS is also known as the Section IV drive shaft.

are attached to the TGB output shaft via two titanium retention plates. At the outboard portion of each paddle assembly, a blade airfoil is attached. The tail rotor blade airfoil is constructed of a fiberglass skin with an aluminum and Nomex honeycomb core. The tail rotor blades are identified by color, and in the order of rotation, as 'red', 'blue', 'yellow', and 'black'.

Tail rotor blade pitch changes are accomplished via movement of the pitch change shaft and pitch beam, which collectively moves the four tail rotor blade PCRs. Each tail rotor blade PCR has elastomeric bearing at their rod ends. A tail rotor servo is mounted to the center housing of the TGB, which provides hydraulic assistance in moving the pitch change shaft and pitch beam.

2.4.2 Tail Rotor Drive System Wreckage Observations

The No. 1 TRDS remained attached to the axial fan via its flexible coupling, but one of its three flange lobes had fractured in overload. The forward end of the No. 1 TRDS had separated from the MGB tail takeoff pinion flange. On the forward flange of the No. 1 TRDS, two of the three flange lobes were present with their attaching hardware, but the third lobe had fractured and separated. No remnant flexible coupling was present on the forward flange of the No. 1 TRDS.

The axial fan and oil cooler radiator were both found separated from each other and from the main fuselage wreckage. The oil cooler radiator lines had fractured and separated from the radiator and the radiator was impact deformed. The axial fan was partially separated from its housing. The axial fan blades were present and exhibited limited curling deformation. The axial fan shaft was found separated from the tail rotor drive train. The forward flange remained attached to the axial fan but had separated from the shaft tube at its riveted connection. The axial fan shaft exhibiting twisting deformation on its aft end in the clockwise direction (aft-looking-forward). The aft end of the axial fan shaft was connected to the No. 2 TRDS's forward flange. The No. 2 TRDS's forward flange had fractured at its lobes, but the remainder of the forward flange remained riveted to the shaft tube. Remnants of the viscous damper support assembly at the aft end of the axial fan shaft was present. The forward portion of the No. 2 TRDS tube was about 30 inches in length and did not exhibit significant twisting deformation but had denting and circumferential scoring on its tube. The aft portion of the No. 2 TRDS was found separate from the main wreckage and its viscous damper (at the aft end of the TRDS) exhibited impact deformation.

The forward end of the No. 3 TRDS had fractured at its flange and the forward flexible coupling exhibited wavy deformation but no fractures. The No. 3 TRDS was continuous and exhibited no twisting deformation. The viscous damper at the aft end of the No. 3 TRDS exhibited no anomalous damage. The No. 4 TRDS exhibited no anomalous damage, was continuous, and connected on both ends to the Nos. 3 and

5 TRDS. The viscous damper at the aft end of the No. 4 TRDS exhibited no anomalous damage. The No. 5 TRDS exhibited no anomalous damage, was continuous, and remained connected to the IGB at its aft end. The IGB remained installed on the tail boom. Continuity of drive was confirmed through the IGB. The IGB oil filler cap remained installed. The No. 6 TRDS was connected to the IGB but had fractured and separated at its upper flange that connects to the TGB. The TGB remained installed on the upper end of the vertical fin, the latter of which had separated from the remainder of the vertical fin. The TGB was whole and did not exhibit fractures of its housing. Continuity of drive was confirmed through the TGB. The TGB oil filler cap remained installed.

2.4.3 Tail Rotor Wreckage Observations

The inboard and outboard retention plates were recovered as an assembly and were still installed on the TGB output shaft (**figure 8**). The 'blue' tail rotor blade was present on the tail rotor assembly but was missing its tip cap. The 'blue' tail rotor blade was whole but exhibited opening of its trailing edge and various impact marks. Remnants of remaining three tail rotor blade spars were present within the retention plate assembly and exhibited a broomstrawed appearance. A 14-inch long piece of a tail rotor blade, composed of spar and afterbody, was found lodged on the right underside of the CRJ700 airplane. (See the Structures Group Chairman's Factual Report for details on this impact location and other structural damage signatures caused by the midair collision.) The remaining three tail rotor blade airfoils were not found.

Figure 8. The recovered tail rotor that remained installed on the TGB. The majority of the 'blue' tail rotor blade was present with the tail rotor wreckage.

2.5 Flight Control System

2.5.1 System Description

The UH-60L has cockpit flight controls for each pilot, composed of a cyclic control, collective control, and a set of pedal controls (**figure 9**). The cockpit flight controls are hydraulically assisted via the boost servos. Inputs from the cockpit flight controls are transmitted to the hydraulic boost servos and mixing unit, both of which are located on the roof on top of the cockpit and covered with a cowling.

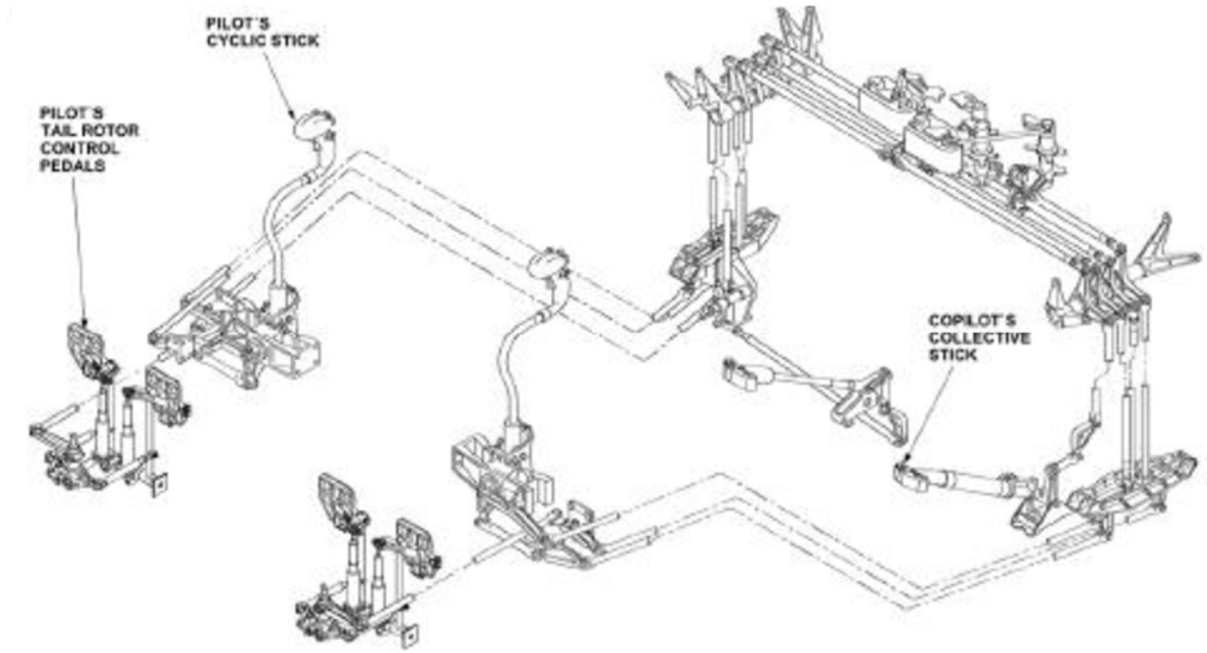

Figure 9. A diagram of the nonrotating flight control linkages, up to but not including the boost servos, of the flight control system. (Image courtesy of Sikorsky)

The No. 1 and No. 2 hydraulic system are independent of each other. Hydraulic pressure for the No. 1 hydraulic system is provided by the hydraulic pump on the left accessory gearbox. Hydraulic pressure for the No. 2 hydraulic system is provided by the hydraulic pump on the right accessory gearbox.

Three hydraulic boost servos are normally powered by the No. 2 hydraulic system and provide hydraulic assistance and damping to the pitch, collective, and yaw axes. Outputs from the mixing unit are transmitted to the main rotor swashplate, via control linkages and bellcranks. Three primary [hydraulic] servos[6], located forward of the MGB and the flight control bellcrank support, provide hydraulic assistance to move the main rotor swashplate and to prevent main rotor feedback loads to the

[6] The left primary servo is "lateral", the center primary servo is "aft", and the right primary servo is "forward".

mixing unit. The primary servos have redundancy via two hydraulic stages, with the first stage powered by the No. 1 hydraulic system and the second stage powered by the No. 2 hydraulic system. The rotating swashplate is driven by two rotating scissor links. Tilting and vertical movement of the swashplate results in main rotor blade cyclic and collective pitch changes via PCRs. The main rotor blade PCRs have elastomeric bearing rod ends. The mixing unit outputs tail rotor pitch blade control commands to the forward cable quadrant. Control cables for tail rotor control are routed from the forward cable quadrant to the aft cable quadrant, the latter of which is located at the tail rotor servo on the TGB.

2.5.2 Nonrotating Flight Controls Wreckage Observations

The left and right pilots' cyclic and collective controls were found in their normally installed location in the cockpit wreckage but exhibited partial separation from the structure. Both the left and right pilots' pedals were found in the cockpit wreckage and were partially separated from the structure.

The flight control linkages from the cockpit controls to the vertical control tubes, and subsequently to the lateral control tubes, were fractured in multiple locations. The left side bellcranks, which transmit the vertical control tube movements to the lateral control tubes, remained attached to the left side vertical control tubes. The right side bellcranks, normally attached between the vertical control tubes (adjacent to the right pilot's seat) and the lateral control tubes, were found separate from the main wreckage; remnant control tubes remained attached to the right side bellcranks. The roll trim actuator, normally installed on the left side of the roof-mounted control linkages, was found within the recovered main wreckage but had separated from its airframe mount. The output link from the roll trim actuator output arm was connected but was fractured at its connection rod to the mixing unit. The yaw trim actuator was found separated from the main wreckage. The output link from the yaw trim actuator output arm remained connected but was fractured at its connection rod.

The boost servo assembly remained installed on remnant structure. The boost servo assembly was whole and did not exhibit fractures or separations of the servos themselves. The collective boost servo connection to the mixing unit was continuous but the link was bent. The collective mixing unit connection had fractured and separated and the mixing unit-end bearing. The pitch boost servo connection to the mixing was intact. The roll boost servo connection was fractured at the output control tube. The yaw boost servo connection had fractured at the output control tube.

The three primary servos remained installed on the airframe (**figure 10**). All three primary servos remained whole and did not exhibit fractures. The control rods on the output side of the right and center primary servos remained connected between the servo and the forward bridge linkages. The control rod on the output

side of the left primary servo had fractured about midpoint of the control rod, but each end remained connected to the primary servo and the forward bridge linkages, respectively. The control rods, including the control rod supports and idler linkages, between the primary servos and the swashplate linkages remained installed around the MGB housing. The input control rods between the mixing unit and left and right primary servos had fractured. The input control rod between the mixing unit and the center primary servo was continuous but the link was bent on its aft end.

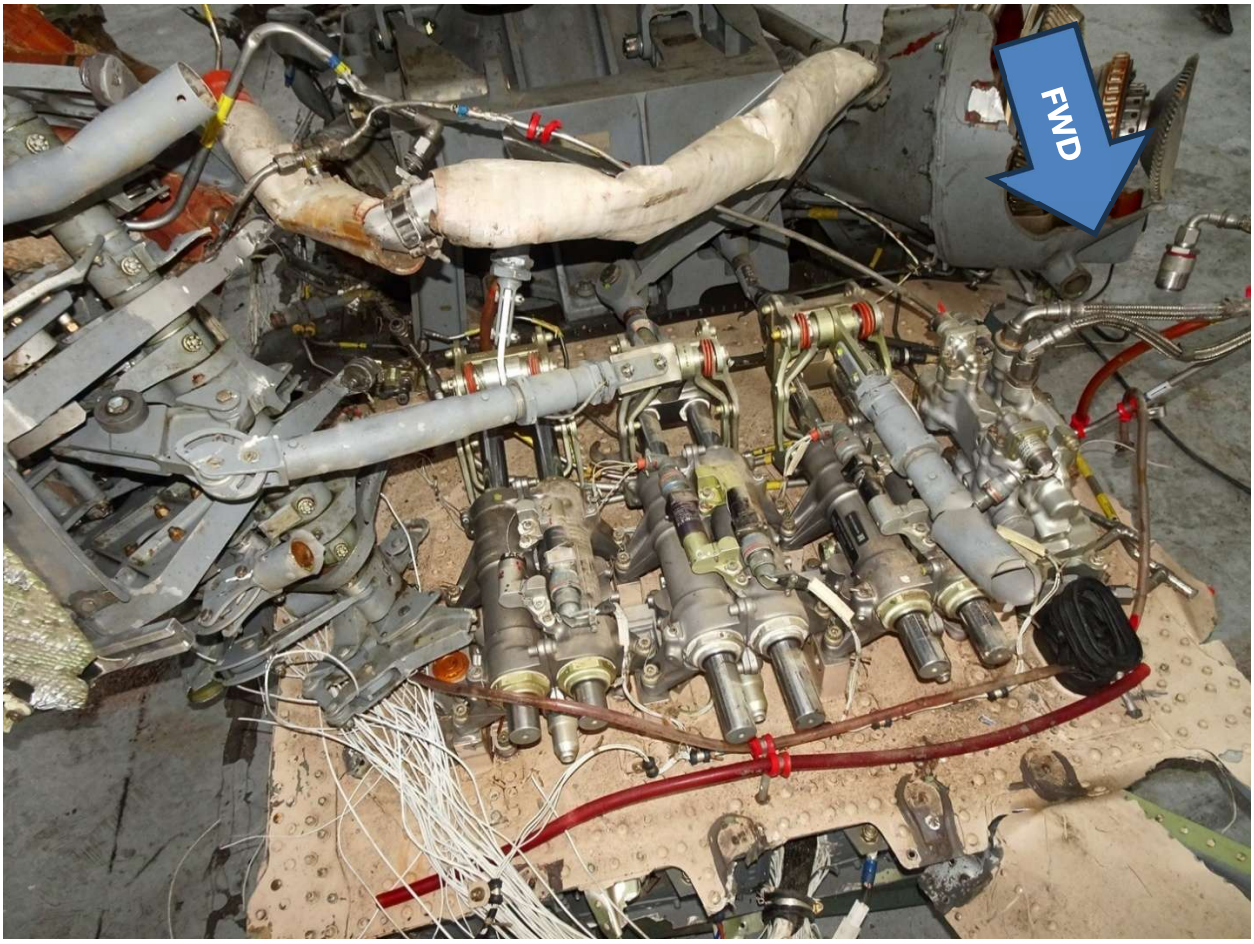

Figure 10. The primary servos installed on the roof structure in front of the MGB.

The tail rotor control cables were not present at their normally installed location at the left and right accessory modules. The tail rotor control cables were fractured at the transition section where the tail boom separated from the main fuselage. The tail rotor control cables were continuous within the tail boom and vertical fin but had fractured in the area where the upper portion of the vertical fin and TGB had separated. Remnant portions of the tail rotor control cables were attached to the tail rotor control quadrant on the TGB. The two spring capsules remained connected to the tail rotor control quadrant, but their mounting bracket had separated.

2.5.3 Main Rotor Rotating Flight Controls Wreckage Observations

The swashplate assembly remained installed on the main rotor shaft. The uniball slider remained installed on the MGB housing. The two rotating scissors remained installed between the rotating swashplate and the main rotor shaft. The swashplate control linkages remained attached to the stationary swashplate. The lower rod ends of all four PCRs remained attached to the rotating swashplate. The 'red' PCR had fractured about 2.5 inches above the lower jam nut. The remainder of the 'red' PCR remained connected to the separated 'red' spindle. The 'yellow' PCR exhibited a straight appearance, was continuous to the upper rod end, and remained connected to its pitch horn. The 'blue' PCR had fractured at its lower rod end bearing. The 'blue' PCR's upper rod end remained connected to its pitch horn, but had fractured about 2.5 inches from the upper jam nut. The remainder of the 'blue' PCR was not found. The 'black" PCR was continuous to the upper rod end, which remained connected to its pitch horn, but was bent about mid-length.

2.5.4 Tail Rotor Rotating Flight Controls Wreckage Observations

The pitch beam and pitch change shaft remained installed on the tail rotor. The 'red', 'yellow', and 'black' tail rotor PCRs were connected on their outboard ends to the pitch beam but had fractured and separated at their inboard rod ends. The inboard rod ends for the 'red', 'yellow', and 'black' tail rotor PCRs were not found. The 'blue' tail rotor PCR remained connected to the 'blue' tail rotor blade pitch horn but was fractured at its outboard rod end. The 'blue' tail rotor PCR outboard rod end remained attached to the pitch beam.

2.5.5 Hydraulic System Wreckage Observations

The right hydraulic pump remained installed on the remnant portion of the right accessory module that was recovered separate from the main wreckage. The right hydraulic pump's lines were fractured and separated from the pump. The forward end of the right hydraulic pump separated, revealing its internal spring. No significant quantity of hydraulic fluid remained within the right hydraulic pump.

The left hydraulic pump remained installed on the left accessory module, but its hydraulic lines were fractured and separated from the pump. The forward end of the left hydraulic pump had separated, revealing its internal spring. No significant quantity of hydraulic fluid remained within the left hydraulic pump.

The backup hydraulic pump and electric motor were recovered separate from the main wreckage. The hydraulic lines were fractured and separated from the backup hydraulic pump. A drain line remained attached to the backup hydraulic pump. The backup hydraulic pump cover was separated, revealing its internal spring. There was no significant hydraulic fluid within the backup hydraulic pump.

The left transfer module remained installed on airframe and its lines to the left hydraulic pump remained installed. The right transfer module was found separated from the airframe but connected via hydraulic lines to the right hydraulic pump. The hydraulic manifold for the primary servos remained installed on the airframe. The hydraulic reservoir for servicing was not present in the recovered wreckage.

2.6 Powerplants

2.6.1 System Description

The accident helicopter was equipped with two GE Aerospace T701-series turboshaft engines. The engines are mounted on the left and right sides of the airframe, behind the main transmission. A hover infrared suppression system (HIRSS) is installed on the exhaust of each engine to reduce the infrared signatures of engine exhaust gases.

The engine control quadrant, located overhead in the cockpit, consists of two power control levers (throttles), two fuel selectors, two engine T-handles (for engine fire suppression), and a rotor brake interlock.

2.6.2 Wreckage Observations

For information on the accident helicopter's engines and auxiliary power unit (APU), see the Powerplants Group Chairman's Factual Report for this investigation. For information on the cockpit engine controls, see the Helicopter Operations Group Chairman's Factual Report for this investigation.

2.7 Air Data System

2.7.1 System Description

The pitot-static system on the UH-60L helicopter is composed of two pitot-static probe assemblies, pitot lines, and static 1 (S1) and static 2 (S2) lines. The pitot, S1, and S2 lines each have drains to allow accumulated water to exit the system. The pitot-static probe assemblies are installed on the left and right sides of the cockpit roof, immediately aft of the cockpit doors (**figure 11**). The probe of each pitot-static probe assembly has two ports for S1 and two ports for S2, with all four static ports located on the outboard side of the probe.

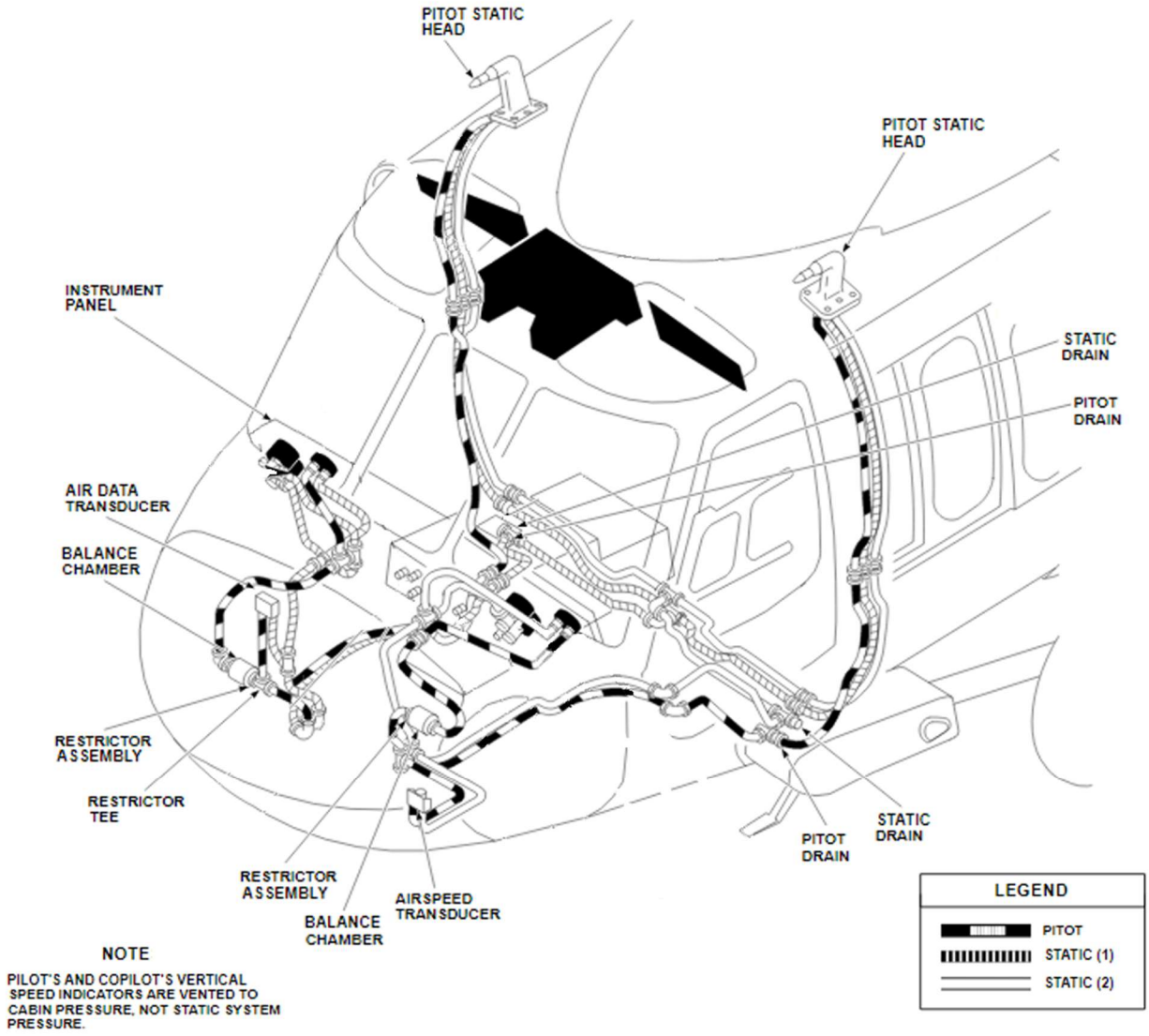

Figure 11. A diagram showing the pitot-static system installation and the pitot, S1, and S2 lines from the probe assemblies to the cockpit instruments. (Image courtesy of the US Army and modified by the NTSB)

Figure 12 shows a schematic of the pitot-static system. The left pitot probe provides stagnation (total) pressure information to the airspeed transducer and to the left pilot's airspeed indicator. The right pitot probe provides stagnation pressure information to the air data transducer, the right pilot's airspeed indicator, and the electronic standby instrument system (ESIS). The S1 lines provide static pressure information to the right pilot's [encoding] barometric altimeter, airspeed indicator, air data transducer, and the ESIS. The air data transducer supplies pressure altitude, indicated airspeed, and altitude rate data to the integrated vehicle health monitoring unit (IVHMU) and the multipurpose flight recorder (MPFR). Information from the air data transducer is consumed by the stability augmentation system (SAS)/flight path stabilization (FPS) computer and the heads-up display (HUD). When the HUD is not in

use, information from the air data transducer is not available in a form that is displayed to either the left or right pilot. The S2 lines provide static pressure information to the left pilot's barometric altimeter, airspeed indicator, and to the airspeed transducer. Information from the airspeed transducer is provided to the SAS/FPS computer and is not available in a form that is displayed to either the left or right pilot.

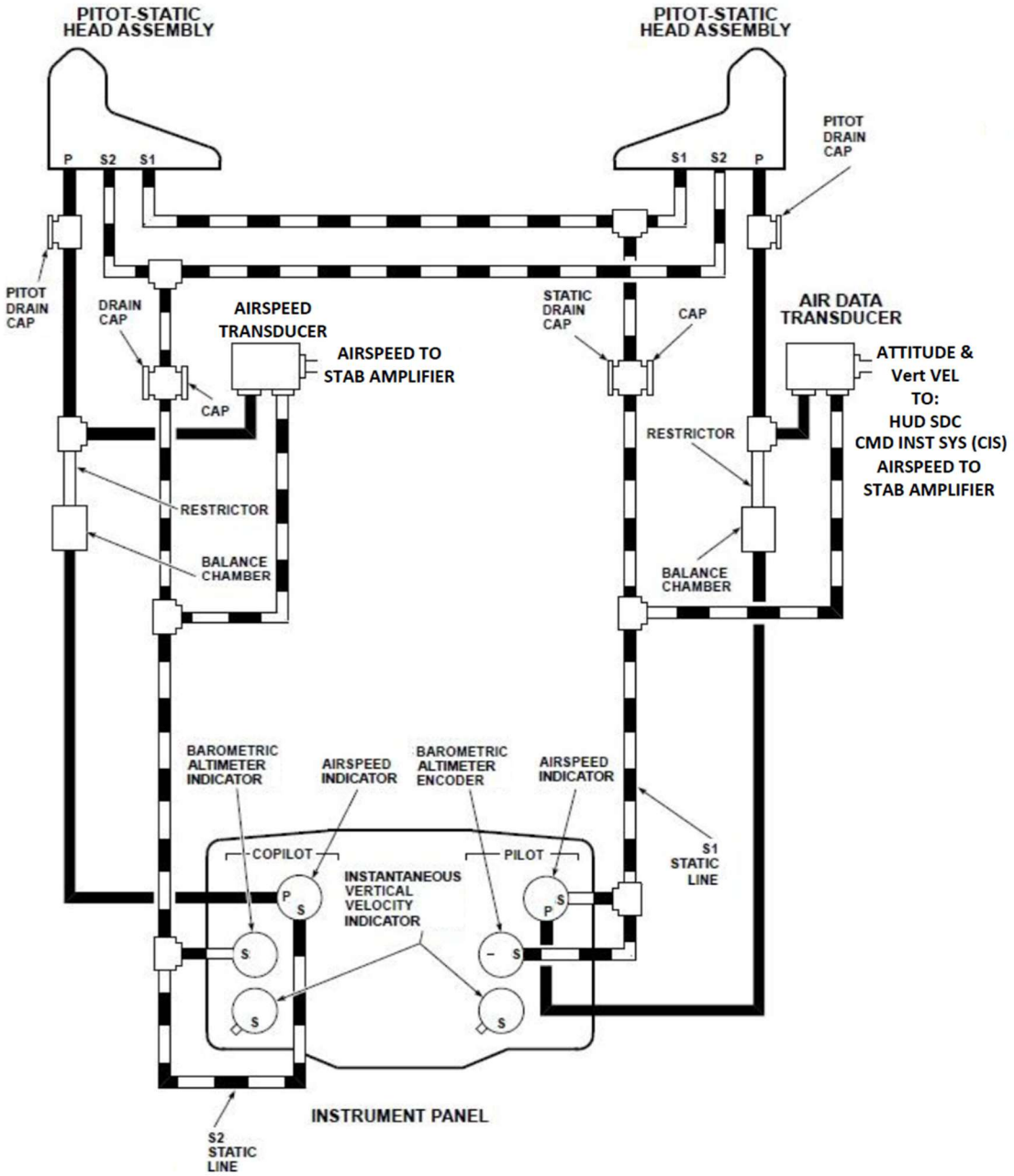

Figure 12. The pitot, S1, and S2 lines that connect to various components in the air data system and the cockpit instruments. The ESIS is not shown in this diagram. (Image courtesy of Sikorsky and modified by the NTSB)

2.7.2 Wreckage Observations

The left and right pitot-static probe assemblies were found separated from the recovered main wreckage (**figure 13**). Both pitot-static probe assemblies contained a small portion of remnant airframe structure attached to them. The probe tip of the left pitot-static probe assembly had fractured and separated; the probe tip was not recovered. The right pitot-static probe assembly contained its probe. The left pitot-static probe assembly showed model 0856HM-7 and S/N 201763. The right pitot-static probe assembly showed model 0856HM-8 and S/N 201168.

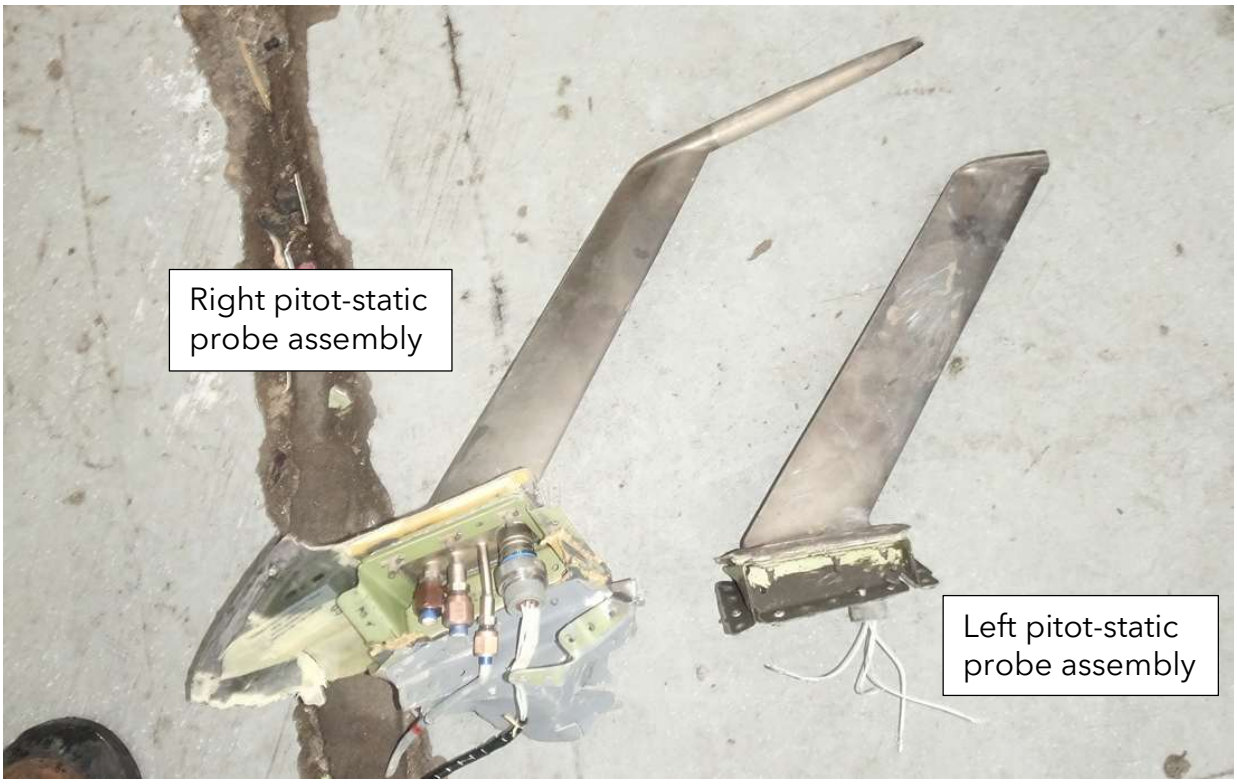

Figure 13. The pitot-static probe assemblies recovered from the Potomac River.

The pitot-static lines exhibited multiple fractures between the location where the two pitot-static probe assemblies are normally installed on the structure to the area of the cockpit instruments. The pitot and static lines from the area where the left and right side pitot-static probes are normally installed were traced along the airframe; they exhibited fractures in multiple locations. The fittings at the unions along these lines were checked for security and all were confirmed to be tight. The fittings on the fractured pieces of the pitot-static system were checked for looseness by hand, and all were confirmed to be tight except for one fitting for a pitot outlet line found near the cockpit nose wreckage. This loose fitting was attached to a T-shaped union that contained a piece of remnant sheet metal as well as another fitting, the latter of which was tight when checked by hand. The loose fitting was examined further by the NTSB Materials Lab in Washington, DC. The lab examination found all

fractures were in overload. The loose fitting and the opposing fitting on the union were both disassembled and exhibited no notable damage features, such as anomalous wear of the threads or the ferrule connection that could indicate looseness and vibratory wear over an extended period of time.[7]

The air data transducer TR-322/A (model 542AY8 and S/N 6797) remained installed in the cockpit nose wreckage (**figure 14**). The pitot and static connectors remained attached to the air data transducer and their connections were tight. The pitot line was fractured about 4 inches from its connection. The static line was continuous, but bent, up to its T-fitting. The three connections to the T-fitting were tight. The static inlet line to the T-fitting was about 13 inches in length but was fractured.

Figure 14. The air data transducer in the cockpit nose wreckage (red arrow).

The airspeed transducer TR-323/A (model 1221 H5 and S/N 6669) was found installed in the cockpit nose wreckage (**figure 15**). The pitot and static lines connected to the airspeed transducer was tight. The pitot lines were continuous to a T-fitting, all of its connections which were tight. A filter remained attached to the T-fitting, but the remaining lines were fractured after the T-fitting. The static line was continuous from the airspeed transducer to a T-fitting and its connections were tight.

[7] Additional details of the NTSB Materials Lab's examination of the loose fitting for a pitot outlet line can be found in **Attachment 1**.

The straight connection into the T-fitting was tight but had fractured at the line. The elbow connection into the T-fitting had fractured at the line and the jam nuts between the elbow and T-fitting exhibited looseness. According to the maintenance manual, this elbow was normally perpendicular to the T-fitting.

Figure 15. The airspeed transducer in the cockpit nose wreckage (red arrow).

2.8 Cockpit Instruments

2.8.1 Barometric Altimeter Description

The left and right pilot's barometric altimeter provides barometric altitude information via an analog pointer. The barometric altimeter setting (Kollsman window), presented in inches of mercury (in-Hg) can be adjusted independently on each altimeter. The left altimeter is model AAU-31/A and the right altimeter-encoder is model AAU-32/A.

The right pilot's barometric altimeter contains an encoding function. The encoding function is a Gillham interface that provides pressure altitude[8] from the

[8] Pressure altitude is the barometrically measured altitude referenced to the standard sea-level atmospheric pressure of 29.92 in-Hg. Equivalently, pressure altitude is the altitude displayed on the barometric altimeter when the altimeter's Kollsman window is set to 29.92 in-Hg, and will differ from

analogue altimeter to the digital transponder, discussed later in this report. This interface uses binary code to provide height information to the transponder in 100 ft increments. The upper-right quadrant of the right pilot's barometric altimeter contains a reddish-orange "OFF" flag that appears when the encoding function is not working.

The design specification for the model AAU-31/A barometric altimeter is controlled by military specification MIL-A-81851B. The design specification for the model AAU-32/A barometric encoding altimeter is controlled by military specification MIL-A-81852A.

2.8.2 Radar Altimeter Description

The radar altimeter set, model AN/APN-209, is composed of two cockpit instruments and two identical, flush-mounted radar altimeter antennas on the underside of the cockpit structure. The left pilot's cockpit instrument is an indicator/receiver-transmitter (IRT). The right pilot's cockpit instrument is only an indicator. One radar altimeter antenna is the transmitter and the other radar altimeter antenna is the receiver. The left IRT also serves to generate and receive the radio frequency signals for the radar altimeter system, and is connected to the two antennas. The left IRT also provides its altitude signal to the right indicator.

Each cockpit instrument has an analog pointer that indicates altitude on a scale from 0 to 1,500 ft as well as an embedded four-digit digital display. On each instrument, a low and high altitude bug can be set, with associated low and high warning lights embedded into each radar altimeter display for when the altitude drops below the low bug or exceeds the high bug. When the radar altimeter system loses power or has an invalid altitude condition, such as a high bank angle or exceeding the altitude limit for the system, an "OFF" flag will appear in the lower-right of each instrument display, covering the digital display.[9]

2.8.3 ESIS Description

The ESIS, model GH-4001, provides the pilots with a visual display of pitch and roll attitude, airspeed, altitude, vertical speed, and heading. The ESIS serves as a backup to the left and right pilot's cockpit instruments and is installed in the center of the instrument panel, between the two pilots' instruments. The ESIS receives static pressure and stagnation pressure information from the S1 and pitot lines, respectively that are connected to the pitot-static system.

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the true altitude when the local altimeter setting differs from 29.92 in-Hg.

[9] According to Honeywell, the manufacturer of the accident radar altimeters, the design specification for the radar altimeter does not require the analog pointer to go behind the mask when the altimeter loses 28V power, but does require that the "OFF" flag is displayed.

2.8.4 Wreckage Observations of the Left Pilot's Instruments

The left pilot's barometric altimeter and horizontal situation indicator were found separate from the main wreckage. The left pilot's barometric altimeter pointer was found near 650 ft and its Kollsman window showed a setting between 29.88 in-Hg and 29.89 in-Hg (**figure 16**). The left pilot's barometric altimeter had its cannon plug attached on its back side, but its static port had fractured and separated and the instrument housing had split on its bottom side (**figure 17**). No data plate was present on the left pilot's barometric altimeter, but an outline of where one was previously installed was visible. The left pilot's radar altimeter, part number (P/N) HG7209AD01 and S/N 0188AA, was found separated from the cockpit wreckage. The left pilot's radar altimeter IRT high altitude bug was found near 800 ft and its low altitude bug was found near 80 ft.[10] The IRT pointer remained installed and indicated near 250 ft, but the IRT glass face had fractured (**figure 18**). The left pilot's airspeed indicator was recovered and its pointer read about 80 knots. The left pilot's stabilator position indicator showed down. The left pilot's vertical situation indicator and vertical speed indicator were found in the cockpit wreckage connected to wiring.

Figure 16. The left pilot's barometric altimeter.

[10] Per US Army TM 1-1520-237-10, page 3-133, the low and high altitude bugs for the radar altimeter are set to 80 ft and 800 ft, respectively.

Figure 17. The separated static port on the back of the left pilot's barometric altimeter. (A blue marker was used by investigators to create an index line on the fitting attached to the static port.)

Figure 18. The left pilot's radar altimeter.

2.8.5 Wreckage Observations of the Right Pilot's Instruments

A portion of the right pilot's instrument cluster was found separated from the cockpit structure but connected via hoses and cables. This right pilot's instrument cluster contained the radar altimeter indicator, barometric altimeter, vertical speed indicator, vertical situation indicator, horizontal situation indicator, airspeed indicator, stabilator position placard, stabilator position indicator, and command instrument system (CIS) mode selector. For the right pilot's radar altimeter indicator, P/N A3060591 and S/N 857AA, its high altitude bug was found near 800 ft and its low altitude bug was found near 80 ft (**figure 19**). The right pilot's radar altimeter indicator pointer was found near 300 ft. For the right pilot's barometric altimeter, P/N 101450-31243 and S/N 173619, its pointer was found near 150 ft and its Kollsman window showed a setting of 29.87 in-Hg (**figure 20**). The right pilot's barometric altimeter had its cannon plug attached on its back side but its static port had fractured and separated from the altimeter housing (**figure 21**). The vertical speed indicator pointer was separated from its dial and was loose but retained within the instrument. The vertical and horizontal situation indicators were not readable. The airspeed indicator pointer was separated from its dial and the pointer was loose but retained within the instrument. This right pilot's instrument cluster was separated from the wreckage by cutting the hoses and cabling on the back side of the instrument cluster.

Figure 19. The right pilot's radar altimeter.

Figure 20. The right pilot's barometric altimeter.

Figure 21. The separated static port on the back of the right pilot's barometric altimeter.

2.8.6 Wreckage Observations of ESIS

The ESIS was found partially separated in the cockpit wreckage and remained attached via wiring (**figure 22**). Its data plate showed part number P/N 9200-45201-0101 and S/N 1267. The glass cover of the ESIS had fractured, but the screen was not cracked. The ESIS was retained for further examination.

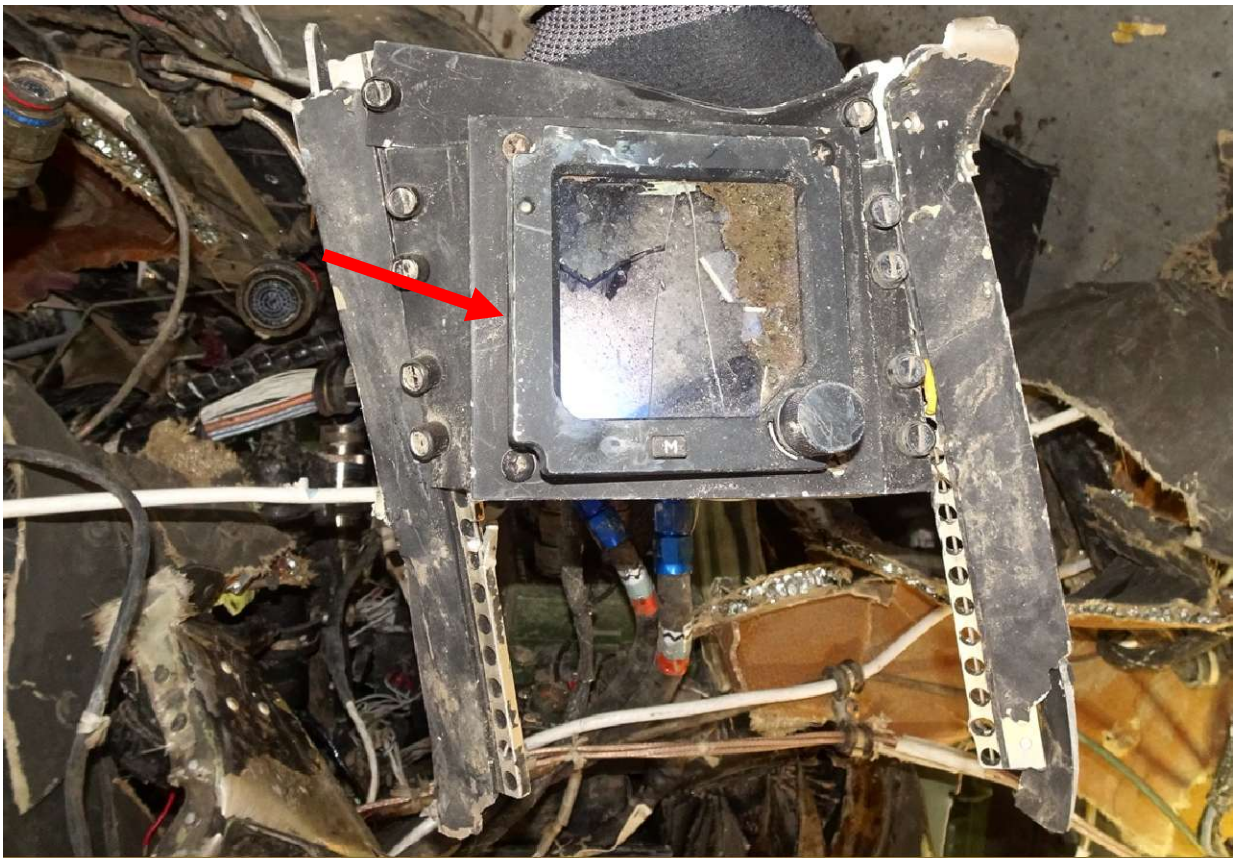

Figure 22. The ESIS (red arrow) as found in the cockpit wreckage.

2.8.7 Radar Altimeter Antennas

The AN/APN-209 radar altimeter right and left antenna were found installed on remnant cockpit wreckage. The right antenna was P/N SM-D-967308A and S/N 1298CC. The left antenna was P/N SM-D-967308A and S/N 0904CC. On both left and right antennae, the connector was attached but the cable was fractured. The remainder of the cabling was continuous through the cockpit nose wreckage but had fractured at the end that normally connects to the left radar altimeter.

2.9 AN/APX-123A Transponder

2.9.1 System Description

US Army UH-60L helicopters are equipped with an AN/APX-123A[11] transponder system, composed of a transponder receiver-transmitter, installed within the nose avionics bay of the helicopter, and a remote control unit (RCU), installed in the cockpit center console. The APX-123A supports Modes 1, 2, 3/A, 5, C, and S. Additionally, the transponder is capable of automatic dependent surveillance-broadcast (ADS-B). The transponder's Mode S function interfaces to traffic alert and collision avoidance system (TCAS) II. The J1 connector cable to the RCU includes wires that are configured to the unique International Civil Aviation Organization (ICAO) aircraft address[12] assigned to that aircraft. Specifically, the wires to pins 12 through 35 of the J1 connector cable to the RCU are configured to either open or ground to program the aircraft address in binary.[13]

The RCU contains a single-line digital display along with a secondary display to indicate enabled mode functions and cautions. The transponder receives GPS position and time information via an interface with an ASN-128D doppler GPS navigation set (DGNS). Additionally, the transponder receives pressure altitude information from the right pilot's encoding barometric altimeter. The encoding barometric altimeter provides pressure altitude information to the transponder in 100 ft increments. Therefore, the pressure altitude reported by the transponder to ground stations, via Modes C and S, as well as ADS-B, were in 100 ft increments.

On the APX-123A transponder, the ADS-B squitter is tied to the Mode S function such that when Mode S is turned off, the ADS-B squitter stops broadcasting. Additionally, the Mode S function can be turned off independently of the transponder. Furthermore, the ADS-B squitter can be turned on or off, independent of the Mode S function. The transponder allows for the Mode S function to be turned on or off while the helicopter is on the ground or while airborne.

2.9.2 Wreckage Observations

The APX-123A transponder receiver-transmitter was found in the cockpit wreckage and remained connected to wiring but had separated from its mount. The transponder receiver-transmitter was P/N A3349792 and S/N 9881. The remote control unit (RCU) for the radar receiver-transmitter was found in the cockpit wreckage with its left dial indicating "A" and the right dial indicating "normal" (**figure 23**). The RCU data label was damaged, precluding identification of P/N and S/N. The

[11] The AN/APX-123A will be referred to as the "APX-123A" for the remainder of this report.
[12] The ICAO aircraft address will be referred to as the "aircraft address" for the remainder of this report.
[13] In this report, a single aircraft address will be expressed in three different ways: hexadecimal, octal, or binary numerals.

two dummy load connectors, associated with the installation of the APX-123A transponder system, remained installed on the airframe underneath the left pilot's seat.

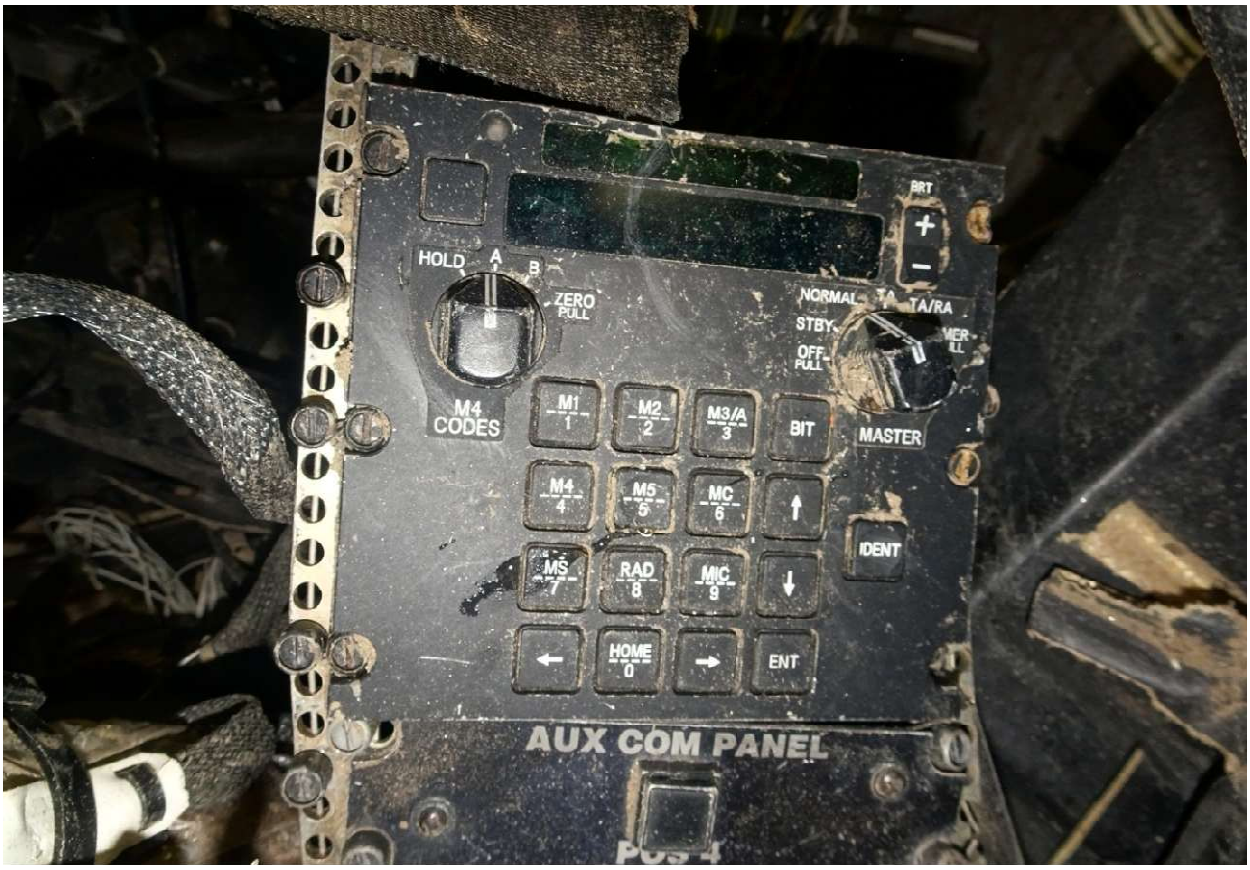

Figure 23. The RCU for the APX-123A transponder found in the cockpit wreckage.

2.10 Miscellaneous Avionics

The ASN-128D DGNS radar signal data converter (SDC) P/N 320A002-04, S/N 3083, and NSN 5895-01-653-0833 was found in the cockpit wreckage. The SDG was separated from its mount and the side of the case housing the connectors had fractured in multiple locations. The wire connectors remained connected to the SDG. The DGNS computer display unit (CDU), P/N P310A022-03 and S/N 0101B, was found in the cockpit wreckage separated from the center console. The wire connectors remained connected to the CDU. The CDU case had fractured at its front face, exposing the internal circuit cards. The DGNS antenna, normally installed underneath the right pilot's seat, was found within, but separated from, the cockpit wreckage.

The accident helicopter was equipped with a MPFR cockpit voice and flight data recorder that was part of the IVHMU, the latter of which was installed by the US Army within the tail transition section. The IVHMU was found in its normally installed

location. An access panel on the underside of the tail transition section was removed to facilitate access to the IVHMU, which was found in its normally installed location with its cable connectors still installed. The IVHMU was removed by divers and transferred to the NTSB for evaluation at the NTSB Recorders Lab in Washington, DC.

The accident helicopter was not equipped, nor was it required to be equipped, with a TCAS.

3.0 ESIS Examination (Model GH-4001)

3.1 ESIS Disassembly

On March 18, 2025, the NTSB Recorders Lab opened the ESIS to extract a circuit card with a nonvolatile memory (NVM) chip of interest. Representatives from L3Harris Aviation Products (L3AP) participated in this activity via video conference. Impact deformation of the ESIS housing prevented normal removal of the housing. Therefore, the ESIS housing was cut along its longitudinal axis using a rotary grinder. The circuit card containing the NVM chip was whole but exhibited deformation on one edge. This circuit card was placed in a vacuum oven and dried for about 4 hours at 35-40° C.

The ESIS wire harness rubber sheathing, at the ESIS connection end, was removed to reveal the data configuration module (DCM). The DCM showed P/N 9230-45005-02 and S/N 0563. The DCM did not exhibit external damage.

3.2 ESIS Data Recovery

On March 31, 2025, representatives from the NTSB and Acron Aviation[14] convened at Acron Aviation facilities in Grand Rapids, MI to recover data from the circuit card and DCM.

The DCM was removed from its parent wire harness assembly through normal means. The DCM connector wires (to its parent wire harness cannon plug) did not exhibit anomalous damage. The DCM was connected to a test box to access its internal memory. The DCM binary file text showed it was a UH-60A and UH-60L configuration.[15] The binary file from the accident DCM was compared to a factory-shipped DCM and did not show substantial differences, indicating the binary file from the accident DCM was likely the same as that shipped from the manufacturer, i.e. the data files were not substantially changed while fielded.

[14] L3Harris' Commercial Aviation Solutions business unit, which included L3AP, was acquired by TJC, a private equity group, and subsequently renamed Acron Aviation.

[15] Normally, the DCM configuration should be the same as that contained in the U16 chip, otherwise a fault will be displayed at power up when the configuration settings between the DCM and the U16 are checked against each other.

The U16 chip, containing the message log, was desoldered from the accident circuit card. Examination of both top and bottom surfaces of the chip showed it appeared in good condition without evidence of cracks. The chip leads were in good condition. The accident U16 chip was subsequently soldered onto a surrogate circuit card on a production unit ESIS (**figure 24**). The capacitor was disconnected from the power circuit card for the purpose of the data download to prevent the normal messages that are recorded to memory being written during a power down. (If there is not enough power from the capacitor to guarantee writing to the message log NVM, the power down messages will not be written to memory.)

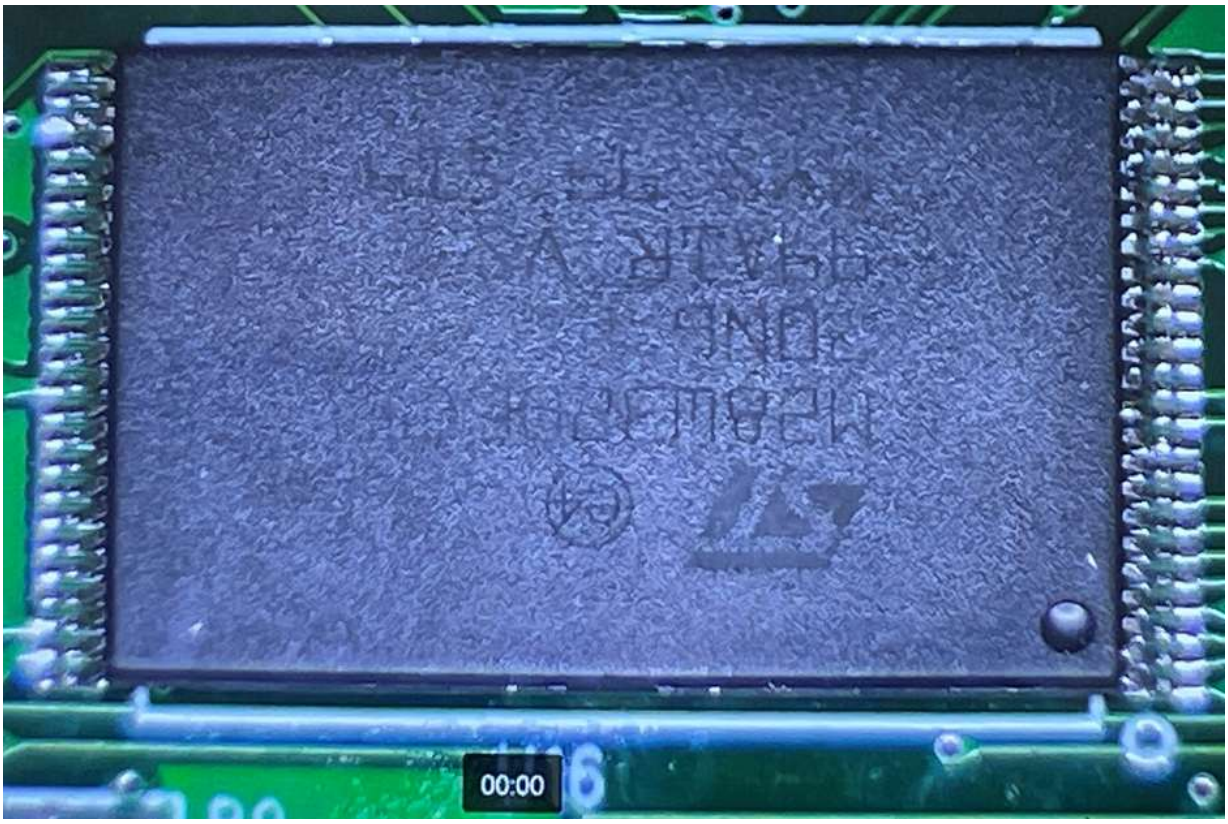

Figure 24. The accident U16 chip after soldering onto a surrogate circuit card. (Photo courtesy of Acron Aviation)

The ESIS was powered up and the display showed a barometric setting of 29.56 in Hg. (Note: This was the barometric setting from the last successful power down write event.[16]) The data onboard was accessed from a test bench (laptop) and the nonvolatile message log showed that block 3 (of 3) was being used to log the most recent flight's messages. (Blocks are written sequentially, therefore in this case block 1 contained the oldest data.)

[16] With the loss of the helicopter's generators, the ESIS is designed to be powered by the backup battery located behind the left pilot's seat.

The most recent data in the message log were multiple faults, such as those from a sensor disconnection, and ended at entry no. 18772.[17] Based on the message log data, the last entry was from a power on time of 1,702.56 hours and a power up count of 1488. In the previous power up count of 1487, the power on time was 1,702.16. In a power up count of 1486, the power on time was 1,701.76 hours. In a power up count of 1485, the power on time was 1,700.12 hours.

In order to determine if the barometric setting logged into the last power up count (1488) was the accident flight itself, the ESIS was powered down. Prior to power down, the ESIS barometric setting was manually changed to 29.60 in Hg. The ESIS was powered up again in order to ascertain the messages that would be created during a new log entry during power up. When the ESIS display was powered up, it showed a barometric setting of 29.56 in Hg, suggesting the changed barometric setting did not save to memory (consistent with the capacitor having been unplugged).

Based on the new data written to the message log, the "level fault" entries were introduced during data recovery for this activity.[18] Therefore, entry 18772 was the last entry and was likely from the accident flight based on the presence of faults that were not found anywhere else in the message log data. Furthermore, the barometric setting of 29.56 in Hg was likely from a flight prior to the accident flight.

The accident U16 chip was left installed on the surrogate circuit card, but the surrogate circuit card was removed from the production unit ESIS. The surrogate circuit card with the accident U16 chip was retained by the NTSB.

According to the manufacturer's records, the accident ESIS was shipped from L3AP on November 2015 and had not returned to L3AP for repair or overhaul.

4.0 Barometric Altimeter Examination

On March 20, 2025, members of the Helicopter Airworthiness Group convened at Aerosonic LLC facilities in Clearwater, Florida (FL) to examine the right and left barometric altimeters from the accident helicopter.

4.1 Right Barometric Altimeter Examination (Model AAU-32/A)

Based on the contract number for the altimeter, found on the data plate on the rear of the altimeter housing, the right barometric altimeter was manufactured in the

[17] The first entry in the message log data, entry no. 1, showed power up count of 1072 and an elapsed time of 1,168.25 hours.
[18] The "level fault" was likely produced due to "attitude fail" displayed on the ESIS due to the accident U16's calibration data not matching with the production unit ESIS.

2002 timeframe. Two labels were affixed on barrel of the altimeter housing. One label showed information that it was overhauled by Olmsted Instrument Company on January 22, 2018. The second label showed the altimeter was certified to 50,000 ft on January 22, 2018 and also showed Olmsted Instrument Company.

The lighting bezel was removed; this part was added by Sikorsky and not delivered by Aerosonic. The barometric setting knob was not present (and was not present at recovery). The altimeter face is normally sealed such that it's only open to atmosphere via the static pressure fitting. With the static port on the accident altimeter having fractured and separated from impact forces, water was allowed to intrude into the altimeter.

The electrical mating plug (on the rear of the unit) was able to be removed normally. Remnant silt was present on the connector surfaces but all pins were present with no visible anomalous damage at the connection. The front glass cover was removed from the altimeter. Moisture was visible on the dial face surfaces.

The gasket between the dial and the front glass cover was present and didn't exhibit cuts or breaks. The dial was removed. Corrosion was seen within the internal metal surfaces. The barometric adjustment gear was removed and its gear teeth and splines exhibited no cracks or fractures. The adjustment knob gearing would not rotate manually.

The rocking shaft pivots were broken, resulting in a drive discontinuity between the aneroid wafer to the gears. Because the rocking shaft pivots were broken, the sector shaft gear's positioning could not be determined to see if it was properly set, i.e. if it was properly engaged to the drive gear. The transmission gear and hair spring gear, normally engaged to each other, were not engaged. The transmission gear normally engages to the drive shaft.

The encoding circuit board for the encoder assembly did not exhibit impact damage or fractures, but two of the pins that connect between the circuit board to the rear circuit board were fractured from the encoding circuit board. The two pins that were fractured were present on the rear circuit board connectors. The encoder circuit board was removed and its connectors did not exhibit anomalous damage. All wiring remained connected. The reader head remained installed with its wiring connected. The encoding function could not be tested as the glass encoding disc was highly fragmented.

Silt and corrosion byproducts were observed on the dial-side of the aneroid wafer, along with a small dent at the edge of the wafer (**figure 25**). This dent was not adjacent to any subcomponents. When rotating the rocking shaft assembly, the sector would not contact, and was not close to, the dent on the dial-side of the aneroid wafer. Another dent was observed on the connector-side of the aneroid

wafer, adjacent to internal structure. The two dents on either side were about 30 degrees apart.

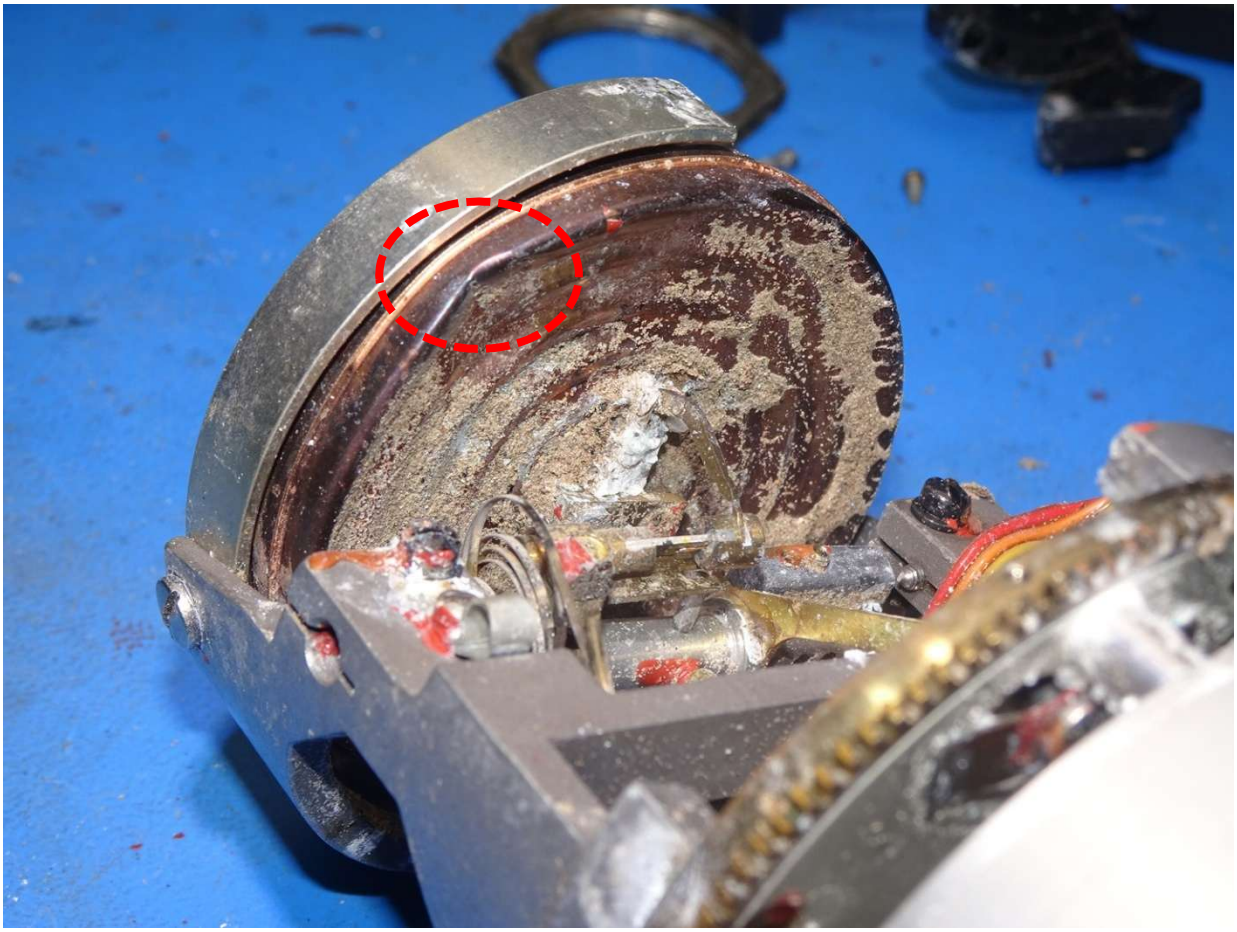

Figure 25. A dent observed on the aneroid wafer on the right barometric altimeter.

The aneroid wafer surface had a "6-18-02" marking in pen, indicating the date it was manufactured. The rocking shaft and slider were removed as an assembly from the aneroid wafer. The hair spring had partially uncoiled. The rocking shaft pivot pins had partially fractured from the housing, resulting in the rocking shaft disengaging from the frame. The center disc holding the encoding glass was present but the encoding glass was highly fragmented; only small, remnant pieces of glass were found within the altimeter. The dial-side pin connecting the gear with the encoding glass disc to the transmission was broken. The jewel (below the gear with the encoding glass) did not exhibit cracking.

The vibrator remained installed. The vibrator was functionally tested by applying power from a test bench. When power was applied, a single movement within the vibrator could be felt, indicative of the vibrator's internal components trying to move. Functional testing was stopped after several power cycles in which the vibrator did not move beyond the single movement.

The gearing from the adjustment (Kollsman window) remained engaged to the output gear and did not show evidence of wear or fractures. The gearing for the Kollsman adjustment barrel could not be rotated freely due to the presence of silt and corrosion on the gear surfaces. The Kollsman adjustment barrel could not be removed from the output gear housing due to seizure of one of the attachment screws; this attachment screw had evidence of corrosion on its head.

4.2 Left Barometric Altimeter Examination (Model AAU-31/A)

The electrical mating plug was removed from the altimeter. The plug pins were all present and did not exhibit damage. No data plate was present on the altimeter.[19] Additionally, it was not known if the altimeter was repaired or overhauled after its original manufacture.

The barometric adjustment knob stem was bent upwards. The adjustment knob was removed from its stem. The glass face was removed; the glass came out separate from the glass housing. (The glass coming out separate of its housing is not considered anomalous during disassembly.) The gasket between the glass housing and the instrument housing was present. The seal between the glass housing to the instrument housing was present.

After the dial face was removed, the internal subcomponents were revealed. A marking in pen was observed, which read "98810 & 32964". The bridge assembly was removed and the driveshaft (for the pointer) that goes through the bridge was removed. There was an engraving on both ends of the bridge assembly; the engraving said "D39".[20] The pinion of the driveshaft was fractured and remained with the bridge assembly. Corrosion was present on the driveshaft splines and gear teeth.

The internal assembly, containing the two aneroid wafers and the two rocking shafts, was removed from the instrument housing. Both aneroid wafers exhibited denting adjacent to the internal structure on the connector-plug side of the wafers. Both rocking shafts were separated from their pivots on both sides. This denting looked similar in appearance to the dent on the right barometric altimeter's aneroid wafer in that it had a well-defined radial indentation (**figure 26**).

The two aneroid wafers were marked by investigators as "I" (left) and "II" (right). Both aneroid wafers showed P/N 202673. The rocking shaft from aneroid wafer I was removed but its pin was not present and not found within the debris. (It is possible

[19] While markings observed on the altimeter's internal components were consistent with Aerosonic markings, the manufacturer of the left barometric altimeter could not be confirmed with the available records.

[20] The practice of engraving the bridge assembly with an identifier was consistent with the work accomplished at the Aerosonic machine shop.

the pin could have separated and departed the instrument since its housing was breached from impact forces.) Rocking shaft I did not exhibit anomalous damage other than surface corrosion. Rocking shaft II was extracted from aneroid wafer II; its pin was still present. The sector on rocking shaft II was bent, but otherwise did not exhibit other anomalous damage aside from surface corrosion. Aneroid wafer II had a dent adjacent to the structure on the dial-side of the wafer, and another dent about 90 degrees away from it (for a total of 3 dents). Both aneroid wafers had pen markings: aneroid wafer I had a "2-28-07" date marking and aneroid wafer II had a "4-4-07" date marking.

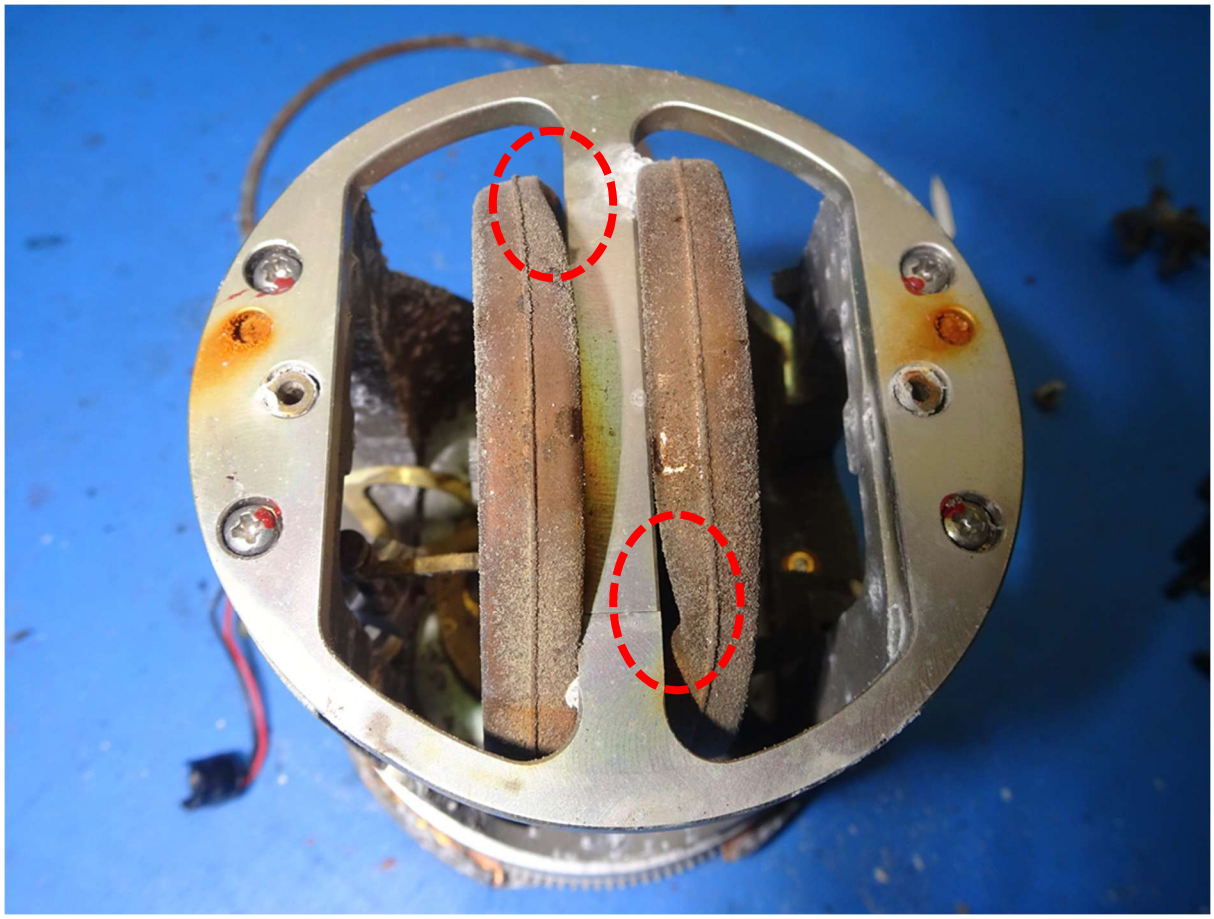

Figure 26. The two aneroid wafers from the left barometric altimeter. The red circles indicate the location of dents observed on these wafers.

The transmission gear was fractured at the pinion end. The gear had corrosion on its surfaces and on its splines. The pivots of the hair spring and gear were fractured. The vibrator was removed and functionally tested. The vibrator was powered but no movement was felt with multiple power cycles. The Kollsman barrel was removed. Continuity of drive was established through the barrel.

4.3 Barometric Altimeter Dial Face and Aneroid Wafer Examination

After completion of the barometric altimeter examination at Aerosonic, the NTSB Materials Lab examined the dial faces of the left and right barometric altimeters to look for evidence of contact marks that may have been left by their pointer, none of which were found.[21] Additionally, lab examination of the dents observed on the aneroid wafers from both the left and right barometric altimeters found no evidence of material transfer onto the dents.

4.4 Barometric Altimeter Tolerances

Military specification MIL-A-81851B governs the performance specifications of the AAU-31/A barometric altimeter, while MIL-A-81852A applies to the AAU-32/A encoding altimeter. Both contain performance tolerances, including allowable scale error tolerances for the altimeter at different pressure altitudes and ambient temperatures. For example, at a standard altitude of 0 ft (an equivalent pressure of 29.921 in-Hg at 0°C), the allowable scale error is ±45 ft at –35°C and ±30 ft at both room temperature and 45°C.

5.0 Air Data System Examination

From April 2-3, 2025, members of the Helicopter Airworthiness Group convened at Collins Aerospace facilities in Burnsville, Minnesota to examine components from the helicopter's air data system. These components included the two pitot-static probe assemblies, the air data transducer, and the airspeed transducer.

5.1 Air Data Transducer Examination (Model 542AY8)

Based on information from the data plate, it was determined that the accident air data transducer was manufactured in November 2000. A review of the manufacturer's records showed the accident air data transducer had not returned to Collins Aerospace for repair or overhaul since the date of manufacture.

The lines for the PS (for S1) and PT (pitot) were removed from their respective ports on the air data transducer housing. There was no significant silt build up seen on the two ports. The electrical connector was removed and the pins appeared intact but there was remnant silt within the connector port. (For more information about the material composition of the silt, see Section 5.3.1, right pitot-static probe assembly, of this report.)

[21] Additional details of the NTSB Materials Lab's examination of the dial face of the altimeters can be found in **Attachment 2**.

The cover for the air data transducer was removed to evaluate its internal condition prior to performing electrical checks. The cover was removed and a significant amount of silt was present within the transducer (**figure 27**). The internal surfaces of the cover did not exhibit evidence of arcing or thermal damage. Dry compressed air and compressed nitrogen were used to clean the silt from the transducer's internal surfaces. Additionally, nylon brushes were used to mechanically clean silt from the internal surfaces. The electromagnetic interference (EMI) box was partially removed from the unit to check for evidence of significant silt deposits, none of which were found within the EMI box. On the accident unit, the epoxy was still intact on the potentiometer resistors' screws, suggesting no changes to the unit calibration after original manufacture.[22]

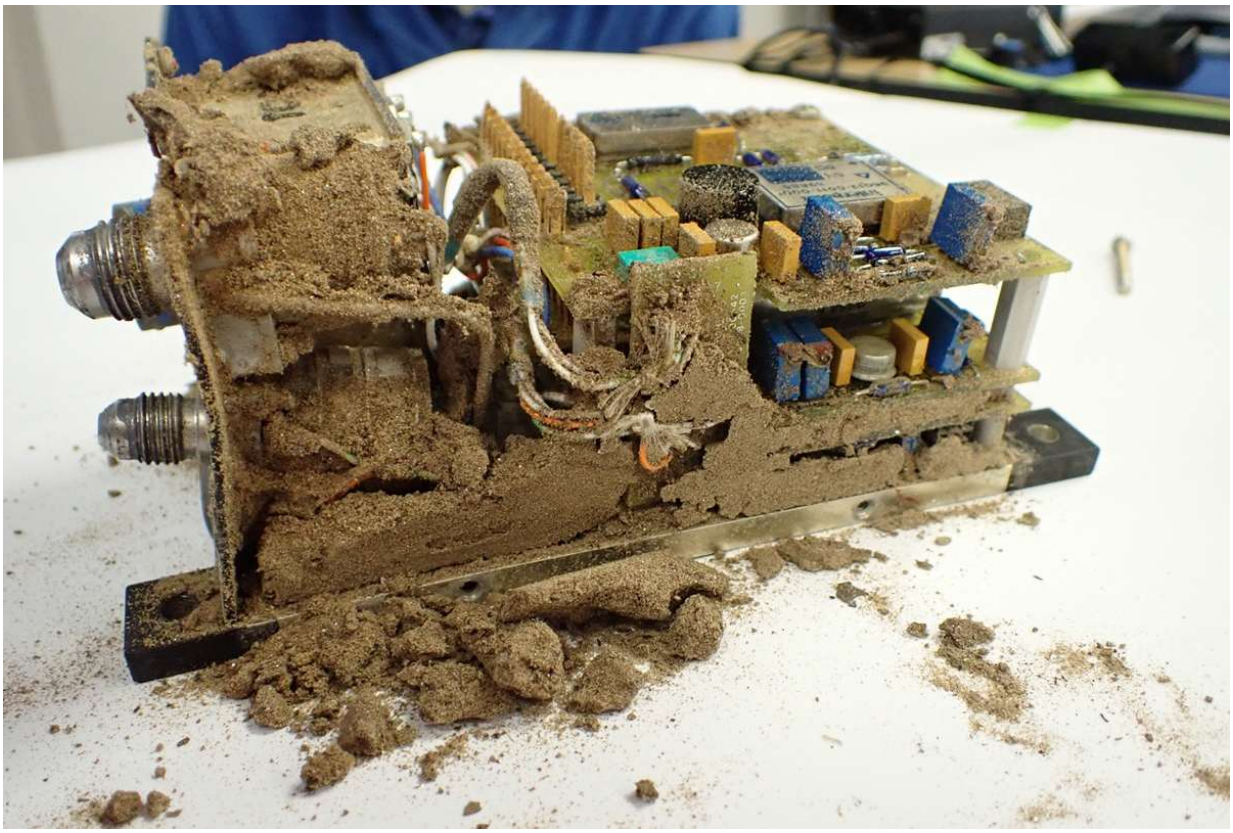

Figure 27. Significant amount of silt found within the air data transducer after removal of its cover. (Image courtesy of Collins Aerospace)

The insulation resistance-to-cable ground (IR) check was performed on a test bench, but the check produced a "fail" result. With the PS (altitude) and Qc (airspeed) sensors disconnected from the circuit card assembly (CCA) stack, the IR check produced a "pass" result. An x-ray of the unit was performed to see if there were contaminants within the sensors that could be a factor in the IR check failure. No

[22] During manufacture of the air data transducer, an epoxy is applied on the potentiometer resistor's screws after final adjustment is accomplished.

contaminants were observed via X-ray within the sensors. The air data transducer was placed in an oven, set to 65° C, for a duration of 1.5 hours to dry any remnant moisture that may be present within the transducer. After drying was completed, another IR check was performed but resulted in a similar result to the first IR check.

The air data transducer was connected to power and the accident sensors remained disconnected. Power was applied with typical power consumption and the unit was powered down in order to connect exemplar sensors to the unit. The unit was subsequently connected to power and powered up again. A high current draw of about 0.5 amps was observed a short time after power application, so power was removed from the unit. Troubleshooting found that the top-most CCA, identified as the 739 CCA (PS sensor and altitude select) was the likely culprit for the high current draw. The CCA stack was removed from the unit and was cleaned using isopropyl alcohol and a nylon brush to remove any residual silt that could affect the CCA stack's proper operation. The CCA stack was reinstalled into the unit and power was again applied, but the current draw remained high at about 0.5 amps. An exemplar 739 CCA was installed onto the accident CCA stack and power was applied to the unit; there was no high current draw. The accident 739 CCA was reinstalled onto the accident CCA stack and power was applied to the unit. A thermal imaging camera was used to identify locations on the transducer that may exhibit high thermal signatures. After several seconds of power application, a small amount of smoke and an acrid smell was observed in the area of the EMI box, after which power was immediately removed. Examination of the EMI box found no significant thermal damage, but the body of one of the feed-through filters had a tan-colored heat tinting. Additional troubleshooting of the accident 739 CCA isolated the U302 (oscillator) chip as the reason for the high current draw.

The U302 chip was replaced with one from an exemplar unit; the oscillator was not anticipated to have impacted calibration of the transducer. An IR check with the sensors still disconnected from the CCA stack produced a "pass" result. An IR check with the sensors re-connected to the CCA stack produced a "fail" result. The accident unit, containing the accident 739 CCA with the donor U302 chip, had sensors from an exemplar unit connected to it, and the unit was powered. The altitude output was first showing about +13V and then showed -10V, representing out-of-range altitude values. The accident 739 CCA was replaced with an exemplar 739 CCA and the accident unit was connected to an exemplar sensor. Power application to the transducer showed the altitude output was stable and within range at about +2V. This +2V was not within tolerance but was expected given the exemplar 739 CCA was paired with a different exemplar sensor, which had not been calibrated to each other. Using the test bench, the altitude input signal was increased, and a resultant corresponding voltage increase of about 10V occurred. Additional troubleshooting, including desoldering of pin 10 from the U102 chip from the 741 CCA, could not determine the voltage anomalies for the 739 CCA. Additional troubleshooting work

on the accident 739 CCA was stopped at this point and the PS sensor was removed from the transducer for individual testing.[23]

The PS sensor was cleaned with isopropyl alcohol, cotton swabs, and a nylon brush in order to remove remnant silt observed on the sensor components (**figure 28**). The PS sensor was placed in a test bench and was tested at two production test points to compare against the original test data from when the PS sensor was manufactured. Additionally, other test points were accomplished that were not production test points. These tests were performed at a temperature of 25°C. (The original test data was at specific pressure points, but did not have defined pressures for altitude from 0-1000 ft at 100 ft increments.) **Table 1** contains the results for this test at 25°C. After the unit was put in an oven and cooled to -15°C, the test was conducted again. **Table 2** contains the results for this additional test at -15°C.

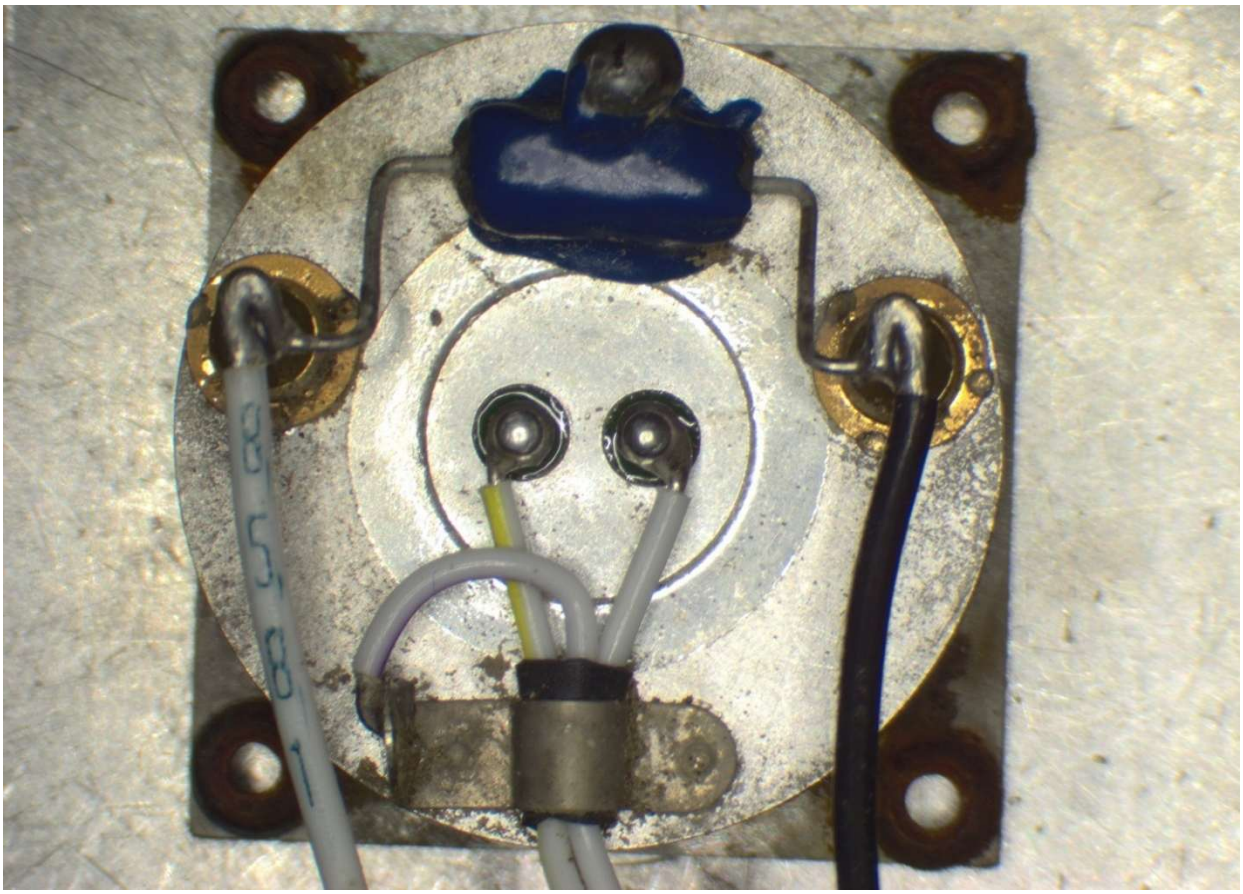

Figure 28. The PS sensor from the air data transducer. (Image courtesy of Collins Aerospace)

[23] At the conclusion of the air data transducer bench testing, the donor U302 chip remained installed on the accident 739 CCA. Additionally, pin 10 of the U102 chip remained de-soldered.

Table 1. Test results from the PS sensor from the air data transducer at 25° C. Cells marked "N/A" indicate no data was collected for these altitude test points. All test results are in picofarads (pF).

Altitude	*Original Test*	*First Test*	*Second Test*
0 ft (14.696psi)	N/A	12.107	12.109
100 ft (14.6429psi)	N/A	12.078	12.080
200 ft (14.590psi)	N/A	12.051	12.051
300 ft (14.5373psi)	N/A	12.023	12.023
400 ft (14.4848psi)	N/A	11.995	N/A
500 ft (14.4324psi)	N/A	11.967	N/A
1,000 ft (14.1726psi)	N/A	11.829	N/A
1,500 ft (13.9166psi)	N/A	10.769	N/A
-2,371 ft (16 psi)	12.9168	12.875	N/A
1,336 ft (14 psi)	11.7683	11.737	N/A

Table 2. Test results from the PS sensor from the air data transducer at -15° C. Cells marked "N/A" indicate no data was collected for these altitude test points.

Altitude	*Original Test*	*Test Result*
0 ft (14.696psi)	N/A	12.076
100 ft (14.6429psi)	N/A	12.047
200 ft (14.590psi)	N/A	12.018
300 ft (14.5373psi)	N/A	11.989
400 ft (14.4848psi)	N/A	11.961
500 ft (14.4324psi)	N/A	11.933
1,000 ft (14.1726psi)	N/A	11.796
1,500 ft (13.9166psi)	N/A	10.738
-2,371 ft (16 psi)	12.8718	12.841
1,336 ft (14 psi)	11.7261	11.707

According to Collins Aerospace, the observed change in capacitance is within demonstrated long term stability of the sensor test system. Generally, the capacitance results between the original data and the test data from this examination showed a decrease in capacitance. For a given pressure altitude, a decrease in capacitance would indicate a higher sensed, and therefore a higher indicated, altitude value. Based on the results of the test comparing the original test result and the current test result, the altitude increase at the 25° C testing would be about +80 ft (average). For the -15° C test, the altitude increase would be about +56 ft (average). These results are for the PS sensor only and does not include the transducer calibration.

5.2 Airspeed Transducer Examination (Model 1221H5)

Based on information from the data plate, it was determined that the accident airspeed transducer was manufactured in June 2000. A review of the manufacturer's

records showed the accident airspeed transducer had not returned to Collins Aerospace for repair or overhaul since the date of manufacture.

The PT and PS (for S2) lines were removed from the airspeed transducer. The PS port contained silt at its opening that was visible once the line was removed. The PT port did not have significant amounts of silt at its opening. The unit's connector pins were in good condition.

X-ray of the unit was performed to evaluate whether the cover could be removed without causing additional damage to the unit. The cover for the airspeed transducer was removed to evaluate the internal condition of the transducer prior to performing electrical checks. Due to impact deformation of the cover, tools were used to pry open and separate the cover from the unit. Significant amounts of silt were found within the unit after the cover was removed. Slight impact deformation was observed on the internal support and a circuit card around the location where the cover was impact deformed. The internal surfaces of the cover did not exhibit evidence of arcing or thermal damage. Compressed nitrogen as well as nylon brushes were used to mechanically clean silt from the internal surfaces.

The airspeed transducer was placed in an oven (65° C for an hour and a half) to dry any remnant moisture that may have been present within the unit. Due to the impact damage to electrical connector, which shifted its position, the test bench's electrical plug could not be installed onto the connector. Three of the four fasteners for the connector were removed and the head of the fourth fastener was cut, allowing for the connector to be freed from the unit to allow for a plug to be connected.

The connector for the IR test would not fit onto the accident unit's connector because the shell of the connector was bent. The accident connector was removed and an exemplar connector was installed in its place. The unit was subsequently hooked up to the test bench and an IR test of the unit produced a "fail" result. Additional work on the airspeed transducer was stopped at this point.

At the conclusion of the airspeed transducer examination, the donor connector remained installed on the transducer.

5.3 Pitot-Static Probe Assembly Examination

5.3.1 Right Pitot-Static Probe Assembly

A review of the manufacturer's records showed the right pitot-static probe assembly was manufactured in September 2000 and had not returned to Collins Aerospace for repair or overhaul since the date of manufacture.

The electrical plug that was secured to the assembly was removed, and the electrical pins appeared in good condition. The three pneumatic connections at the bottom of the probe assembly were identified as pitot (forward), S1 (middle), and S2 (aft). The S2 port contained remnant silt. On the probe head, the lower of the two S1 and S2 ports both had foreign material, which had the appearance of silt, clogging these ports. The upper of the two S1 and S2 port were both clear of foreign material.

Throughout the surface of the probe head and the leading edge area of the strut, a foreign material that was rough to the touch was present. The foreign material was adhered more tenaciously than the silt remnants observed on various components. A sample of foreign material adhered to the probe head was removed and its elemental composition was checked via energy-dispersive X-ray spectroscopy (EDS) in the scanning electron microscope (SEM). The EDS spectra showed peaks predominantly of calcium (42.83%), oxygen (29.96%), and fluorine (10.12%). Silt samples were also checked via EDS in the SEM. The EDS spectra for these silt samples predominantly showed peaks of silicon and oxygen, though a small amount of calcium was seen in these samples.

X-ray of the pitot-static probe assembly revealed no evidence of blockages within the pitot, S1, and S2 passageways.[24] A small void was observed on the leading edge of the strut. The void had a dark-colored appearance consistent with a filled material (such as braze) as opposed to a crack void, which would appear light in color in the X-ray.[25]

Pressure leak testing of the pitot-static probe assembly was performed. The pitot, S1, and S2 lines passed the pressure test of 40 psi of pressure for 1 minute. IR testing of the probe heater produced a "fail" result. However, the IR values observed were high enough such that the heating function of the probe was anticipated to work properly if power is applied to it.

Borescope inspection of the pitot and S1 lines and the S2 strut volume found no evidence of obstructions. Small amounts of remnant silt were observed sporadically within the pitot and S1 lines and the internal surfaces of the S2 strut volume. The silt-like material was removed from the S2 port; by this point the silt-like material on the S1 port had already cleared itself. Subsequently, compressed air was applied into the pitot, S1, and S2 lines to clear any remnant silt prior to installation of the probe into the wind tunnel (**figure 29**).

[24] The pitot and S1 ports on the probe head are directly connected to the corresponding fittings on the baseplate by a pneumatic tube. The S2 ports on the prob head are connected to the strut cavity which has an opening in the baseplate connected to the S2 fitting.
[25] According to Collins Aerospace, the void was within acceptable manufacturing tolerances and was likely present since the manufacture of the right pitot-static probe assembly.

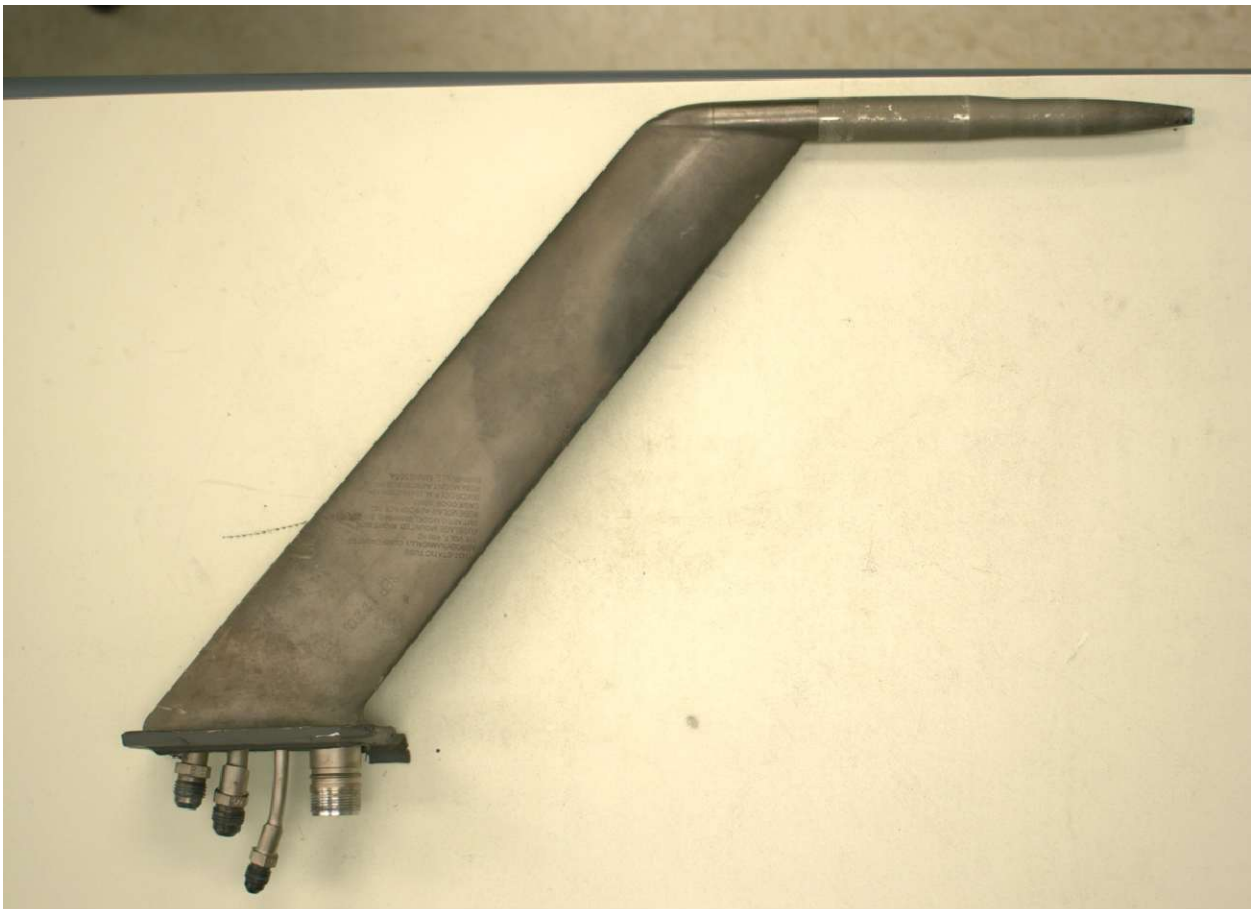

Figure 29. The right pitot-static probe after removal of remnant structure and fittings but prior to wind tunnel testing. (Image courtesy of Collins Aerospace)

The standard (a calibrated exemplar) for the 0856HM-8 pitot-static probe assembly was tested in the wind tunnel to acquire a baseline for comparative purposes. The range of speeds was 60-120 knots. Subsequently, the accident right pitot-static probe was installed in the wind tunnel and tested. Wind tunnel testing showed the readings deviated slightly from the standard, with the maximum altitude error being about 3 ft. **Figure 30** shows the difference in the wind tunnel testing results between the standard and the accident pitot-static probe assembly.

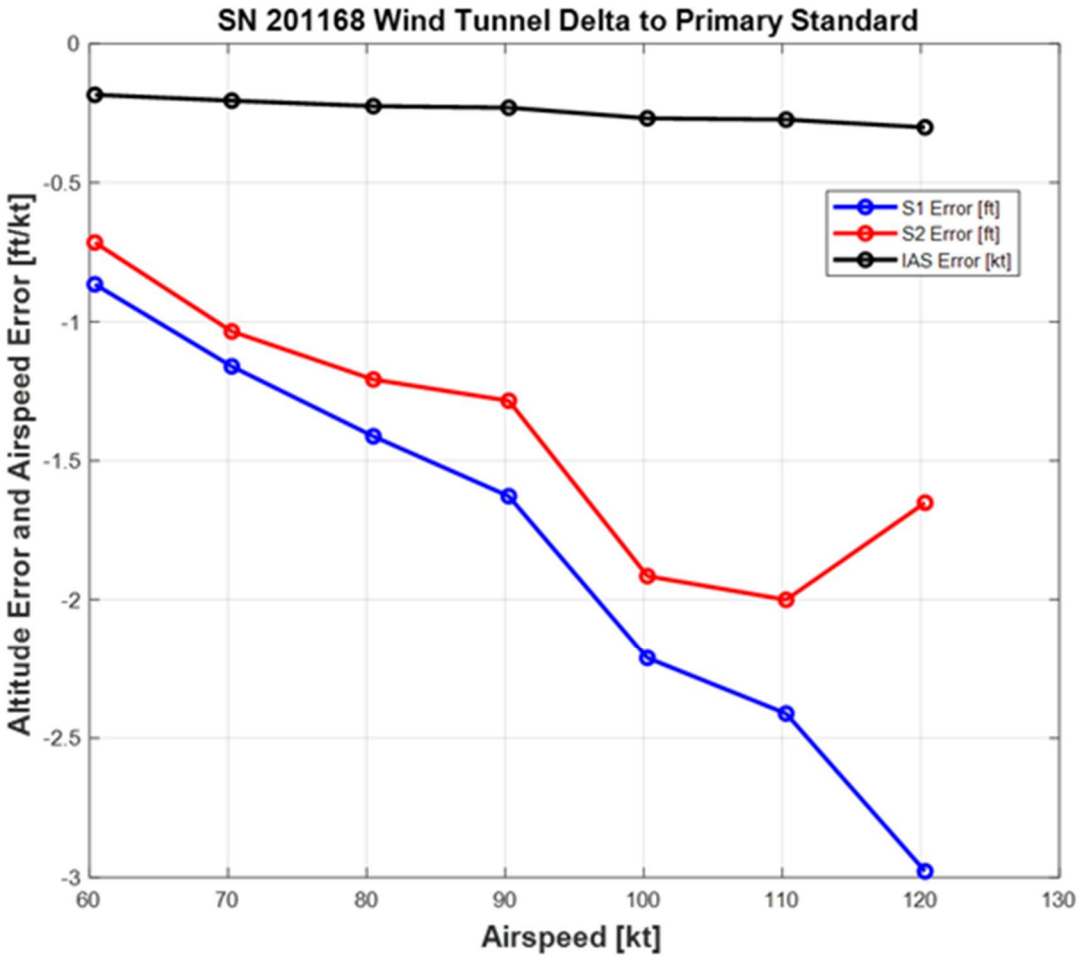

Figure 30. The S1, S2, and airspeed error from the wind tunnel testing calculated between the standard and right pitot-static probe assembly. (Image courtesy of Collins Aerospace)

5.3.2 Left Pitot-Static Probe Assembly

A review of the manufacturer's records showed the left pitot-static probe assembly was manufactured in October 2000 and had not returned to Collins Aerospace for repair or overhaul since the date of manufacture.

The electrical connector was secured to the assembly and was removed. The pins appeared in good condition. The three pneumatic connections were pitot (forward), S1 (middle), and S2 (aft). X-ray of the probe assembly revealed no evidence of blockages within the pitot, S1, and S2 passageways. The calcium deposits were not seen on the left probe's strut surfaces.

The probe head heater pin-to-strut was checked with a multimeter and showed it was shorted and pin-to-pin was open. The probe strut heater pins were checked with a multimeter and showed continuity.

Borescope inspection of the pitot and S1 lines and the S2 area found no evidence of obstructions. Small amounts of remnant silt was observed sporadically within the pitot and S1 lines and the internal surfaces of the S2 strut area, though comparatively less than the right pitot-static probe assembly.

5.4 Conceptual Static System Error Calculation

On the UH-60L, the SAS/FPS computer uses data from both the air data transducer and the airspeed transducer. At an indicated airspeed above 60 knots, the SAS/FPS computer will produce a cockpit indication to the flight crew if it detects an indicated airspeed difference greater than 25 knots between the airspeed information received from the air data transducer and from the airspeed transducer. This indication includes illumination of the "FLT PATH STAB" caution light on the caution panel.

Using this indicated airspeed difference, the NTSB requested Collins Aerospace calculate a conceptual pressure altitude error band that could occur at the indicated airspeed difference threshold of 25 knots. Assumptions made for this calculation are: 1) the primary error contributor for the indicated airspeed difference was the static pressure measurement of the airspeed transducer, independent of the cause of the static pressure measurement error, and 2) the helicopter's flight condition was the altitude, rounded to 300 ft, and airspeed, about 73 knots, at the time of the midair collision. Based on these conditions, with an indicated airspeed difference of 73 knots and 48 knots, the calculated pressure altitude difference between the two static sources supplied to the transducers was about 135 ft. With an indicated airspeed difference of 73 knots and 98 knots, the calculated pressure altitude difference between the two static sources supplied to the transducers was about 193 ft.

6.0 Radar Altimeter Examination (AN/APN-209)

From April 8-9, 2025, members of the Helicopter Airworthiness Group convened at Honeywell facilities in Clearwater, FL to examine the radar altimeters and their antennas.

6.1 Left Radar Altimeter IRT

The left IRT's dial pointer remained at about 250 ft. The low altitude bug remained at 80 ft and the high altitude bug remained at 800 ft. The "OFF" flag was in view. The analog zero adjustment cover was found on the case adjacent to the analog zero adjustment potentiometer but not covering the analog zero adjustment

potentiometer. Remnants of the covers for the digital zero adjustment potentiometer and the volume potentiometer were present. The four connector plugs on the back side of the IRT had fractured and separated, precluding any testing of the IRT.

The heads of several screws retaining the housing to the instrument had fractured and separated, but the remaining screws were removed. The housing had to be pried off. The altimeter was removed, and remnant silt was present on the circuit card assembly surfaces. Impact damage on two of the circuit cards was present and co-located with denting on the housing. The rear panel was removed. Neither the rear panel nor the housing's internal surfaces showed evidence of thermal damage or arcing. The circuit card assembly stack was disassembled. Visual evaluation of the circuit cards showed no evidence of missing components and no thermal damage or arcing.

The receiver and transmitter monolithic microwave integrated circuit (MMIC) modules remained installed on the IRT. On the receiver, the TNC[26] connector had fractured at its two flanges and separated from the receiver. The two flanges and its two screws remained installed on the receiver. On the transmitter, the TNC connector had fractured at its two flanges and separated from the transmitter. One of the flanges and its screw remained installed on the transmitter, but the other flange and its screw head were separated and not found.

The indicator portion of the IRT was partially disassembled and no anomalous damage was found internally beyond corrosion (**figure 26**). The indicator pointer rotor was able to be turned manually but exhibited roughness and binding initially, but the movement freed up with additional manual movement. Power was applied to the indicator motor, and it turned initially clockwise (looking at its face), but subsequent application of power did not result in additional movement. No additional testing of the IRT was conducted.

[26] TNC is an acronym of "Threaded Neill-Concelman".

Figure 26. The left radar altimeter indicator after removal from the IRT housing. (Image courtesy of Honeywell)

6.2 Right Radar Altimeter Indicator

The right indicator's dial pointer remained close to the 300 ft mark. The low altitude bug remained at 80 ft and the high altitude bug remained near the 800 ft mark. The "OFF" flag was in view. The aircraft plug at the J1 connector remained installed on the back side of the altimeter indicator. The NVG bezel and its wire remained installed on the cover of the indicator. The connector covers over the J2 connector remained installed on the back side of the indicator, which is normal for this installation on the UH-60L.

The aircraft-side plug at the J1 connector was removed and all pins were present and appeared to be in good condition with the exception of surface corrosion. The housing was deformed inward in the area of a circuit card, preventing normal removal of the housing.

The altimeter was connected to a test bench. The indicator was powered with 5V and the green-colored instrument illumination light came on. 28V was applied but the "OFF" flag still was in view and the dial pointer would not move with altitude changes commanded through the test bench. The indicator was subsequently removed from the test bench.

The low altitude bug adjustment knob was manually straightened to support removal of the bezel. The bezel was removed. Both the low and high altitude bug adjustment knobs were manually manipulated which resulted in movement of their respective bugs along the perimeter of the dial face. The housing had to be cut in order to separate the housing from the rest of the indicator. The indicator was placed on the test bench and had similar results to the initial test in which the green-colored instrument lights illuminated, but dial pointer did not move.

The power supply board was removed, and the remainder of the indicator was placed on a test bench. Power was applied to the unit and the green-colored instrument lights came on (**figure 27**). The "OFF" flag stowed with application of power, and the "OFF" flag was in view when power was removed. The right set knob, which contains a "push to test" function when pressed, was pressed, and the bench PTT light illuminated, indicating continuity from the PTT button to a lower level circuit card. Movement of the test bench's altitude adjustment dial resulted in sudden and erratic movement of the pointer. The left set knob was rotated to position the low bug to the proximate 6 o'clock position, which engaged the off microswitch. Engagement of the off microswitch resulted in the "OFF" flag correctly coming into view and illumination of the test bench's "off" bulb, confirming functionality of the off microswitch. The LO and HI lights did not illuminate. The four digit digital indicator did not show numbers and had a tan color. (The left IRT four digit digital indicator was a green color.) With variable altitude inputs, the voltage per foot scaling showed fairly consistent correlation to the specification requirements between 0 to 150 ft, with the pointer indicating near, but slightly below, its required altitude with known voltages. However, after 150 ft, the pointer immediately moved to above 1500 ft. Based on the results of this functional testing, additional disassembly of the unit was not performed.

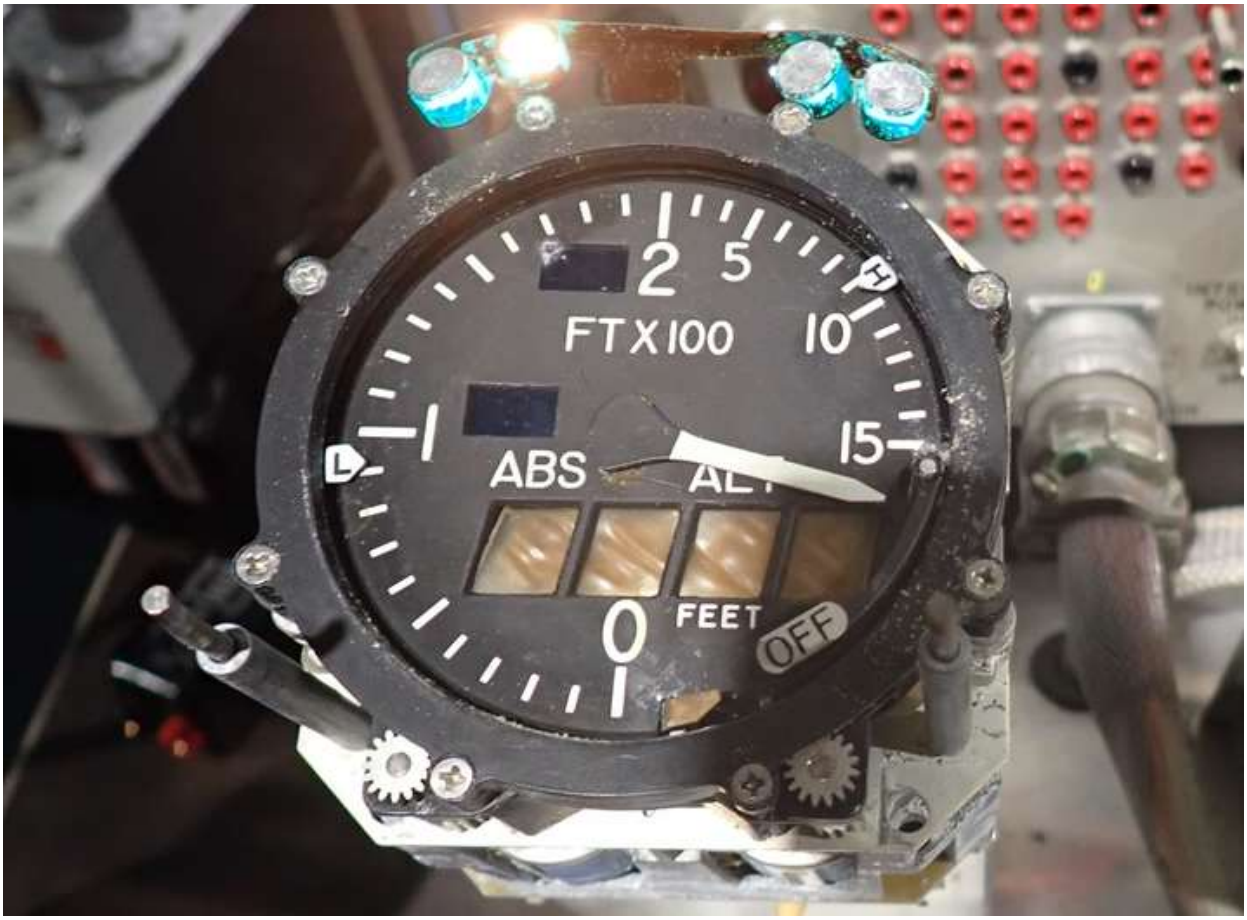

Figure 27. The right radar altimeter indicator during bench testing. Green-colored instrument lights are illuminated above the dial face and the "OFF" flag is stowed. (Image courtesy of Honeywell)

6.3 Left Radar Altimeter Antenna

The left radar altimeter exhibited a slight deformation and exhibited delamination on its edges. As designed, the antenna is normally flat in form. The left antenna was plugged into a vector network analyzer (VNA) and placed into an anechoic chamber. The voltage standing wave ratio (VSWR) was compared to the manufacturer's requirement. The VNA was swept from 4.25 GHz to 4.35 GHz range (the assigned frequency band for radar altimeters). Two of the resultant numbers were outside of the manufacturer's specification but the third result (the upper-side band) was within limit; see **table 3**.

Table 3. The VSWR results for the VNA frequencies tested.

Frequency (GHz)	*VSWR*
4.25	4.09
4.3	2.81
4.35	1.58

6.4 Right Radar Altimeter Antenna

The right antenna exhibited more deformation compared to the left antenna. The dielectric (middle) and antenna array (outer) layers had completely separated and were not recovered. The right antenna was plugged into a VNA and placed into an anechoic chamber. The VNA was swept from 4.25 GHz to 4.35 GHz range (the assigned frequency band for radar altimeters). **Table 4** shows the resultant VSWR numbers exceeded the manufacturer's specification.

Table 4. The VSWR results for the VNA frequencies tested.

Frequency (GHz)	*VSWR*
4.25	74
4.3	59
4.35	47

6.5 Radar Altimeter Dial Face Examination

After completion of the radar altimeter set examination at Honeywell, the NTSB Materials Lab examined the dial faces of the left radar altimeter IRT and right radar altimeter indicator to look for evidence of contact marks that may have been left by the pointer, none of which was found.[27]

7.0 Transponder System Examination (Model APX-123A)

From April 15-16, 2025, members of the Helicopter Airworthiness Group convened at BAE Systems facilities in Greenlawn, New York to examine components from the accident helicopter's transponder system.

7.1 RCU Examination

The RCU assembly's label was damaged and precluded identification of information on it. The back plate remained installed on the unit and the J1 connector did not exhibit looseness. The J1 connector pins appeared to be in good condition. Remnant silt was present throughout the J1 connector surfaces. Pin 24, which is part of the inputs that determines the aircraft address discretes, was identified. Using a plastic tool, pin 24 was gently pushed and did not show evidence of looseness.

The back plate's screws were removed, revealing the input/output (I/O) circuit cards and wire bundles within. The wire connections between the power supply card and the I/O circuit card were intact. Limited corrosion damage was observed on capacitors on the power supply card. Within the housing of the RCU, there was a

[27] Additional details of the NTSB Materials Lab's examination of the dial face of the altimeters can be found in **Attachment 2**.

label that showed "60271-4 MSN1804". A data tag on the I/O circuit card showed "11263 ASSY 69617-1 REV:D S/N 2287".

The connector for the input wires to the power supply cable (from the J1 connector) was removed and showed limited corrosion on its surfaces. The J2 connector on the power supply card was removed; it showed also limited corrosion on its surfaces. On the inner side of the J1 connector, a black-colored shrink wrap sleeve for the wire for pin 24 appeared to look higher than adjacent wires' shrink wrap sleeves, exposing the solder cup for the pin 24 wire (**figure 28**). The wire for pin 24 easily came out from its J1 connector cup with light finger pressure; the wire had separated from the connector cup and remnant solder was present on the wire.[28] The cup for the pin 24 wire, to which the wire is soldered, had white-colored material on its exterior surface. The adjacent unused cups, to which no wires were soldered, had a gold-colored appearance with comparatively less white-colored material on their surfaces. The other white-colored wires on the J1 connector were tugged with needle nose pliers, but none of them separated from their cups.

The connectors from the power supply card to the I/O circuit card were removed. The I/O circuit card showed limited corrosion on its surfaces and connector pins. The corrosion was removed using isopropyl alcohol, brushes, and cotton swabs. A black light inspection of the I/O circuit card showed the conformal coating was present and in good condition throughout the circuit card.

[28] The damaged wire connection to pin 24 is relevant to the Mode S address read by the RCU. See Section 7.2 of this report for more information on the Mode S address for the accident helicopter.

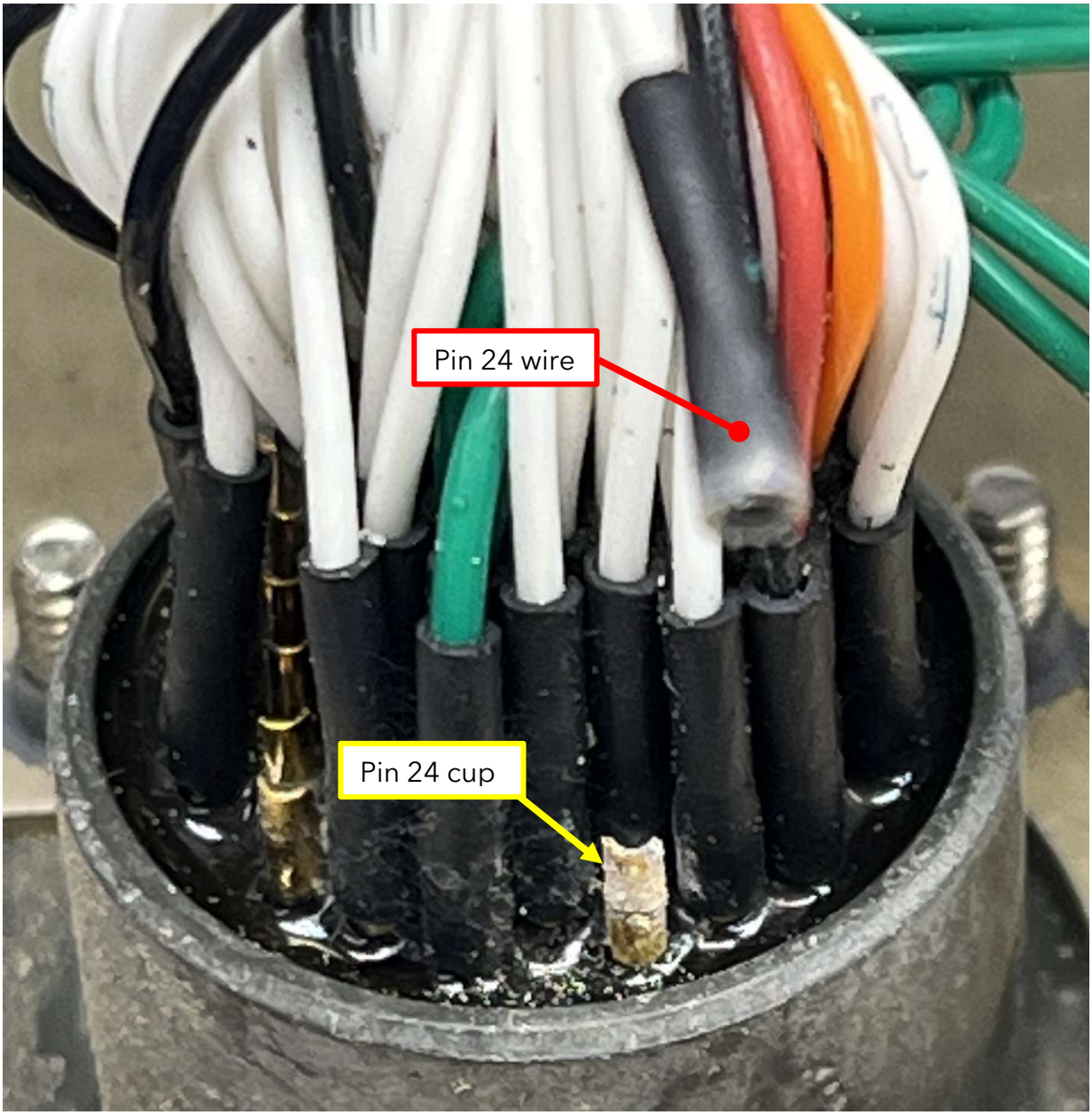

Figure 28. The inner side of the J1 connector of the RCU. The yellow arrow points to the cup for pin 24. The red arrow points to the wire that was connected to the cup for pin 24. The pin 24 wire is in the foreground and was separated from the cup. (Image courtesy of BAE Systems)

On an exemplar RCU, the exemplar I/O circuit card was removed and the accident I/O circuit card was installed in order to access the accident I/O circuit card's NVM that contained the transponder settings for the accident flight. The exemplar RCU, containing the accident I/O circuit card, was connected to a test bench and the test bench was configured in an "in-flight" configuration, i.e. weight-on-wheels

discrete was off.[29] Power was applied to the RCU and the RCU booted up. The RCU showed Modes 3/A, C, and S were illuminated (enabled). Going through the RCU menus, the following settings of interest were noted (**figures 29 and 30**):

- ADS SQTR OFF[30]
- MS DATA ON
- ID PAT25
- M3/A 0125
- ADDR AE313D[31]
- TIME SRC 153[32]
- RCUSWVER 5G

Figure 29. The ADS-B squitter setting.
(Image courtesy of BAE Systems)

Figure 30. The time source setting.
(Image courtesy of BAE Systems)

[29] When the APX-123A RCU is powered during an in-flight configuration, the RCU will retain previous operating settings. Conversely, when the RCU is powered when in an on-the-ground condition, the RCU will erase certain settings. Furthermore, when the RCU is powered when in an on-the-ground condition, the RCU will read the hardwired aircraft address from the J1 connector cable and populate the aircraft address field in the RCU with this hardwired address. Subsequently, the aircraft address can be manually changed on the RCU via its menus and the number pad keys.
[30] When ADS SQTR is "OFF", the ADS-B squitter is set not to broadcast.
[31] This setting is the aircraft address that was populated in the RCU for the accident flight. See Section 7.2 of this report for more information on the aircraft address for the accident helicopter.
[32] The time source setting is used to control the time and position data source that is used by the RCU. See Section 8.1 of this report for more information on the configuration of the APX-123A transponder after installation on US Army UH-60L helicopters.

When the number pad keys were pressed, there was no consistency in the operating force needed to push the keys. The 3, 6, and 7 number keys were easier to push than the 4 number key, ident, and right keys. The control knob covers were removed to facilitate removal of the front panel. The front panel was removed. Rust-colored corrosion was seen at the base of the right knob. Corrosion was seen at the base of the left knob. Silt and rust-colored by-product were seen on the back surface of the switchboard card. A gasket was present between the RCU housing and the front face plate. The switchboard card was separated from the face plate. White-colored corrosion was seen in limited areas at the edges of the switchboard card. On the back side of the display's translucent cover, remnant silt was present. On the back side of the face plate, a label showed the following:

NSN: 5935-01-619-9911
PN: 5-230-003
REV. –
MFR 6ELP9
SUN DISPLAY SYSTEMS LLC

A date label on the back side of the face plate showed "DATE: 05-15" and another label showed "S/N: 1248".

The buttons on the switchboard card did not show anomalous damage and its springs were present. A label on the switchboard circuit card showed "11263 ASSY 69615-1 REV: C S/N 2180".

The switchboard card was cleaned using brushes, cotton swabs, and isopropyl alcohol. The switchboard card was reinstalled on the accident face plate, which was subsequently reinstalled on the accident RCU housing. The accident I/O circuit card was reinstalled on the accident RCU housing. The power supply card and J1 connector from the exemplar RCU was installed on the accident RCU assembly for the purpose of bench testing. The accident RCU assembly was connected to the test bench using the same settings as the previous power up, and the test bench was powered. The accident RCU assembly booted up successfully. The display showed the same data in the RCU menus as during the previous power up, including Mode S showing as an active mode. Pressing the number 7 key for Mode S resulted in the "S" extinguishing on the RCU display; this is expected behavior when pressing the 7 key. Subsequently pressing the number 7 key again turned illuminated "S" on the RCU display. Power was removed from the test bench and work was concluded on the accident RCU assembly.

7.2 RCU J1 Connector Cable Examination

The accident helicopter was assigned a hexadecimal aircraft address of AE213D (octal 53420475). However, the aircraft address transmitted during the

accident flight was AE313D (octal 53430475). Additional information about the transmitted aircraft address can be found in Section 9.0 of this report. These two aircraft addresses, expressed in binary, are seen in **table 5**, with the pin position for the J1 connector cable to the RCU shown for each binary digit.[33] Pin 24 contains the different binary digit needed to change the aircraft address from AE213D to AE313D.

Table 5. The binary expressions for the two hexadecimal aircraft addresses, AE213D (octal 53420475) and AE313D (octal 53430475). Pin 24 is shown in bold and red-colored text to highlight the single difference between these two aircraft addresses.

Aircraft Address	Pin No.											
	35	34	33	32	31	30	29	28	27	26	25	24
AE213D	1	0	1	0	1	1	1	0	0	0	1	**0**
AE313D	1	0	1	0	1	1	1	0	0	0	1	**1**
Aircraft Address	**Pin No.**											
	23	**22**	**21**	**20**	**19**	**18**	**17**	**16**	**15**	**14**	**13**	**12**
AE213D	0	0	0	1	0	0	1	1	1	1	0	1
AE313D	0	0	0	1	0	0	1	1	1	1	0	1

The aircraft address configured on the accident helicopter's J1 connector cable [to the RCU], specifically pins 12 through 35, was checked using a multimeter. The results of the binary aircraft address that was read from the accident helicopter's J1 connector cable can be seen in **table 6**. The binary aircraft address configured for the accident J1 connector cable was consistent with the assigned aircraft address for the accident helicopter.

Table 6. The binary aircraft address that was read from the accident helicopter RCU's J1 connector cable.

Pin No.											
35	**34**	**33**	**32**	**31**	**30**	**29**	**28**	**27**	**26**	**25**	**24**
1	0	1	0	1	1	1	0	0	0	1	0
Pin No.											
23	**22**	**21**	**20**	**19**	**18**	**17**	**16**	**15**	**14**	**13**	**12**
0	0	0	1	0	0	1	1	1	1	0	1

7.3 NTSB Materials Lab Examination of the RCU J1 Connector

After completion of the transponder RCU examination at BAE Systems, the NTSB Materials Lab examined the separated wire for pin 24 and the J1 connector of the RCU.[34] The cup for pin 24 had no wire remnants or bulk solder present. Examination of the solder at the end of the wire for pin 24 found its surface was

[33] The binary aircraft address is read in descending order from pin 35 to pin 12.
[34] Additional details of the NTSB Materials Lab's examination of the soldered wires to the J1 connector of the RCU can be found in **Attachment 3**.

mostly smooth with wear streaks, consistent with an incomplete bond with the cup's internal surface. Pull testing on adjacent wires found that the wire itself would neck and fracture prior to separation of their soldered joint, with remnant wire still affixed to the cup via solder. However, the wire for pin 31 separated from its cup at the soldered joint when pulled, without any wire remnants or bulk solder remaining on the cup for pin 31. Visual examination of the solder at the end of the wire for pin 31 found its surfaces to have a smooth appearance, similar to that of the wire and cup for pin 24. Neither cup for pin 24 and pin 31 exhibited evidence of fractures of the cup itself.

8.0 Accident Helicopter Equipment, Maintenance, and Inspections

According to the accident helicopter's logbook, the helicopter had an aircraft total time (ATT) of 4,803.6 hours at the start of the day of the accident (January 29, 2025).

8.1 Equipment Installed on Tail No. 00-26860

8.1.1 Transponder and DGNS

According to the accident helicopter's aircraft logbook, the APX-123A transponder was installed in accordance with modification work order (MWO) 1-1520-237-50-118 on April 25, 2023.[35] The aircraft logbook showed the transponder, radar receiver-transmitter RT-1912A(C) was P/N A3349792 and S/N 9881. According to the aircraft logbook, the transponder was installed at a "nomenclature installed hours"[36] of 4,445 hours, a "component install hours"[37] of 1 hour. The aircraft logbook did not show a P/N and S/N for the transponder RCU.

According to MWO 1-1520-237-50-118, a prerequisite for installing the APX-123A transponder was for MWO 1-1520-237-50-105 ("*Installation of the AN/APX-123 MODE 5 Common Transponder*") and MWO 1-1520-237-50-117 ("*Installation of the AN/ASN-128D Doppler GPS Navigation Set 2*") to be applied before or concurrently with the application of MWO 1-1520-237-50-118.

According to the accident helicopter's aircraft logbook, the ASN-128D DGNS was installed in accordance with MWO 1-1520-237-50-117 on April 25, 2023. The

[35] MWO 1-1520-237-50-118 is titled "*Modification Instructions for the Improved Digital Transponder (CXP), AN/APX-123A and Automatic Dependent Surveillance-Broadcast (ADS-B) Out Capability on the UH-60A/L Aircraft*." According to this MWO, the purpose was "to install the improved digital transponder set AN/APX-123A Common Transponder (CXP) and ADS-B Out capability to meet a Federal Aviation Administration (FAA) mandate on UH-60A/L aircraft. This MWO listed an effective date of April 18, 2019 and a completion date of April 18, 2024.

[36] "Nomenclature installed hours" is equivalent to the ATT at the time of component installation.

[37] "Component install hours" is equivalent to the component total time at the time of component installation.

SDC for the ASN-128D, P/N P320A002-04 and S/N 3083, was installed at an ATT of 4,463 hours and a component total time of 40 hours.

According to the accident helicopter's aircraft logbook, MWO 1-1520-237-50-105 for the installation of the APX-123 transponder was issued on June 15, 2012 with a compliance deadline date of April 20, 2019. According to the accident helicopter records, MWO 1-1520-237-50-105 was accomplished on September 25, 2013.

8.1.2 Integrated Vehicle Health Monitoring System (IVHMS)

The IVHMS is a system composed of various sensors, a remote data concentrator, and the IVHMU. The IVHMU collects accelerometer data for various dynamic components for health trending purposes and to identify components that may exhibit anomalous accelerations that may indicate degradation or imminent failure. The MPFR is incorporated within the IVHMU.

According to the aircraft logbook, the IVHMS was installed on the accident helicopter on April 2, 2015 in accordance with MWO 1-1520-237-50-100.

8.2 Scheduled Inspections

The accident helicopter had accumulated an ATT of 4,803.6 hours as of the morning of January 29, 2025. US Army technical manual (TM) 1-1520-237-20&P, titled *Field Maintenance Manual Including Repair Parts and Special Tools List for Helicopters, Utility Tactical Transport UH-60A and UH-60L*, dated 31 May 2024, contains criteria for scheduled and conditional inspections as well as procedures to accomplish these inspections.

8.2.1 Preventative Maintenance Daily (PMD) Inspection

The PMD inspection contains inspection requirements for the last flight of the day, before the first flight of the next mission day, or seven calendar days. The PMD inspection required, but was not limited to, the following items:

- Check exterior for general condition and obvious damage.
- Inspect antennas for evidence of lighting strikes, damage, cleanliness, and security.
- Check [tail pylon] anti-collision light, antenna, and mount for general condition and security.

The last PMD inspection was accomplished on January 28, 2025 at an ATT of 4,803.6 hours.

8.2.2 Preventative Maintenance Service (PMS)

The PMS is accomplished every 40 hours. The PMS required, but was not limited to, the following items:

- Inspection of the pitot-static tubes for obstructions, such as dirt and insects.
- Inspection of the pitot-static tubes for discoloration beyond that normally caused by the pitot heater.

On the accident helicopter, the PMS was last accomplished on October 8, 2024 at an ATT of 4,769.9 hours. **Attachment 4** contains an excerpt from the accident helicopter's logbook showing accomplishment of this PMS.

8.2.3 Phase Maintenance Inspection (PMI)

PMIs are major inspections that are separated into two distinct inspections: PMI-1 and PMI-2. Every 480-hours, a PMI is accomplished, with the PMI-1 occurring first, followed by the PMI-2, after which the next PMI inspection is a PMI-1. The PMI-1 inspection required, but was not limited to, the following items:

- Perform an operational check of the pitot tube heaters.
- Perform an operational check of the pitot-static system.
- Perform an operational check of communications equipment.
    - #1 FM Radio Set Clear Voice/ICS Backup Tuning Check.
    - UHF Radio Set Clear Voice Check.
    - VHF Radio Set Clear Voice and Backup Control/ICS Backup Tuning Checks.
    - #2 FM Radio Set Clear Voice/ICS Backup Tuning Check.
    - SATCOM.
- Perform an operational check of the radar altimeter.
- Perform an operational check of anti-collision lights.
- Perform an operational check of position lights.
- Perform an operational check of formation lights.
- Perform an operational check of utility and maintenance lights.
- Perform an operational check of the controllable searchlight.
- Perform an operational check of the retractable landing light.
- Perform an operational check of cargo hook lights.

For the accident helicopter, the last PMI-1 was accomplished, in conjunction with the last PMS, on October 8, 2024 at Fort Liberty[38], North Carolina at an ATT of

[38] On February 14, 2025, Fort Liberty was renamed to Fort Bragg.

4,769.9 hours. Additionally, the last PMI-2 was accomplished on July 13, 2022 at Fort Belvoir, VA at an ATT of 4294.7 hours. **Attachment 4** contains excerpts from the accident helicopter's logbook showing accomplishment of these PMIs.

After accomplishment of a PMI, a maintenance test flight (MTF) is necessary under the requirements of TM 1-1520-237-MTF.[39] The US Army provided NTSB investigators with a copy of TM 1-1520-237-MTF dated April 30, 2023. Within the "before starting engines" section of the MTF procedures, step 28 stated:

> *28. Barometric altimeters – Check tower setting against setting for field elevation. Unreliable for flight if more than 70-foot error exists.*[40]

According to accident helicopter records, the last MTF was accomplished at Fort Liberty, North Carolina on October 17, 2024 at an ATT of 4,773.6 hours.

8.2.4 Pitot-Static System Scheduled Inspections

The 12-month inspection, work package (WP) 0291, contains instructions for inspecting the pitot-static system. This inspection requires removal of the caps from the pitot and static drain ports from underneath the pilot's and copilot's seats to drain water from the lines, coating the caps with antiseize compound, installing the caps, and an operational check of the pitot-static system to include the altimeter, vertical velocity indicator, and the airspeed indicator per WP 0098.

The operational check of the pitot-static system, per WP 0098, uses a pitot-static system test setup that attaches to the pitot-static head assembly to simulate different total and static pressures. The test checks the operation of the left and right pilots' barometric altimeters, vertical speed indicators, airspeed indicators, and the barometric altimeter lighted bezel assembly. The test set altitude indicator is initially set to 0 ft and the test set barometric scale at 29.92 in-Hg. **Table 7** shows the results that are required to be observed on the left and right pilot's altimeters for different test set altitudes (without change to the barometric scale).

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[39] The US Army defines a MTF as "a flight for which the primary mission is to determine airworthiness, i.e. that the airframe, power plant accessories and items of equipment are functioning in accordance with predetermined requirements in the intended operational environment."

[40] In the FAA Aeronautical Information Manual (AIM), Chapter 7 ("Safety of Flight") Section 2, titled *Barometric Altimeter Errors and Setting Procedures* states that "Manufacturing and installation specifications, along with 14 CFR part 43, Appendix E requirement for periodic tests and inspections, helps reduce mechanical, elastic, temperature, and installation errors." Furthermore, paragraph 7-2-3 states that "If the difference from the known field elevation and the altitude read from the altimeter is plus or minus 75 feet or greater, the accuracy of the altimeter is questionable and the problem should be referred to an appropriately rated repair station for evaluation and possible correction."

A review of an exemplar pitot-static system test kit used by the 12th BN found a pitot and static system tester, model no. ADTS405 and manufactured by Druck Ltd., and an air data accessories kit, model no. ADA60A-835 and manufactured by Nav-Aids Ltd.

Table 7. The test set altitude and barometric altimeter requirements for the operational check of the pitot-static system

Test set altitude	*Barometric altimeter shall indicate:*
0 ft	-50 ft to 50 ft
500 ft	470 ft to 530 ft
1,000 ft	965 ft to 1,035 ft
2,500 ft	2,430 ft to 2,570 ft
5,000 ft	4,920 ft to 5,080 ft
10,000 ft	9,900 ft to 10,100 ft

According to the aircraft logbook, the last 12-month pitot-static system inspection was accomplished on May 8, 2024 at an ATT of 4,724.1 hours. **Attachment 5** contains excerpts from the accident helicopter's aircraft logbook showing accomplishment of the 12-month pitot-static system inspection.

8.2.5 APX-123A Transponder Scheduled Inspections

MWO 1-1520-237-50-118 required that from the installation date of the APX-123A transponder, a 12-month battery replacement is required. According to the accident aircraft logbook, the APX-123A transponder battery was last replaced on May 16, 2024 at an ATT of 4,724.1 hours.

8.2.6 IVHMS Scheduled Inspections

The PCMCIA[41] card within the data transfer unit, located at the aft end of the cockpit center console, records data from the IVHMS. This PCMCIA card is removed at the end of each flight and downloaded to a ground station to determine if any vibration levels exceed established thresholds.

At a 120-hour interval, the IVHMS data is reviewed further for trending purposes. This 120-hour task was last accomplished on October 22, 2024 at an ATT of 4,774.7 hours. This task does not require validation of the parametric data that is recorded by the MPFR.

[41] PCMCIA acronym is "personal computer memory card international association".

8.3 Recently Reported Discrepancies of Interest in the Aircraft Logbook

8.3.1 Pitot-Static System Discrepancies

On September 13, 2024, at an ATT of 4,769.9 hours, a discrepancy was noted in which the pitot-static system failed initial checks. The corrective action, dated September 23, 2024, noted that the pitot-static system attachments were tightened and an operational check, without aircraft power applied, resulted in an acceptable result. The corrective action did not specify which attachments on the pitot-static system were tightened. **Attachment 6** contains this discrepancy entry for the pitot-static system.

8.3.2 Exterior Lighting System Discrepancies

On July 18, 2024, at an ATT of 4,742 hours, a discrepancy was noted in which the upper anti-collision light electrical connector required cleaning due to the anti-collision light intermittently being on and off. This discrepancy was closed as completed on the same day.

On July 11, 2024, at an ATT of 4,732.7 hours, a discrepancy was noted in which the upper anti-collision light would not illuminate in day or night mode. This discrepancy was closed the same day, with the corrective action stating that a maintenance operational check (MOC) was "ok".

On May 20, 2024, at an ATT of 4,724.1 hours, a discrepancy was noted in which the lower anti-collision light electrical connector jack was found loose. The corrective action, dated June 6, 2024, stated the lower anti-collision light was replaced.

On April 18, 2024, at an ATT of 4,708.5 hours, a discrepancy was noted in which the tail position light was inoperative, with a power cycle attempt not resolving the issue. The corrective action, dated the same day, stated that the fault was entered in error.

9.0 Accident Helicopter ADS-B and Transponder Aircraft Address

This section discusses the ADS-B function of the accident helicopter's transponder as well as the transponders installed on other UH-60-series helicopters at the 12th BN. The 12th BN had several Stratus 2S portable wireless ADS-B receivers that could receive ADS-B data and wirelessly connect to a tablet to display that ADS-B information. Discussion of the US Army's policies and procedures for the use of the APX-123A transponder's functions, including ADS-B, as well as use of the Stratus 2S can be found in the Helicopter Operations Group Chairman's Factual Report.

9.1 Accident Helicopter ADS-B History

For the accident flight, the accident helicopter's transponder began to respond to Mode S interrogations near Cabin John, Maryland. The accident helicopter's transponder continued to respond to Mode S interrogations until the midair collision occurred. For the accident flight, Mode S multilateration (MLAT) data was populated in publicly available ADS-B databases, but ADS-B data was not transmitted by the accident helicopter (**figure 32**).

On the night of the accident flight, there was no Mode S data from the accident helicopter prior to the start of Mode S transmissions near Cabin John. On the night of the accident flight, radar stations tracked the accident helicopter via Modes 3/A and C from its departure at DAA and subsequently via Mode S when the helicopter was near Cabin John. These radar stations included DCA, Dulles International Airport (IAD), Baltimore/Washington International Thurgood Marshall Airport (BWI), and Joint Base Andrews (ADW). These radar stations tracked other aircraft in the area, and no other aircraft transitioned from Modes 3/A and C to Mode S during the same timeframe that the accident helicopter began responding to Mode S interrogations.

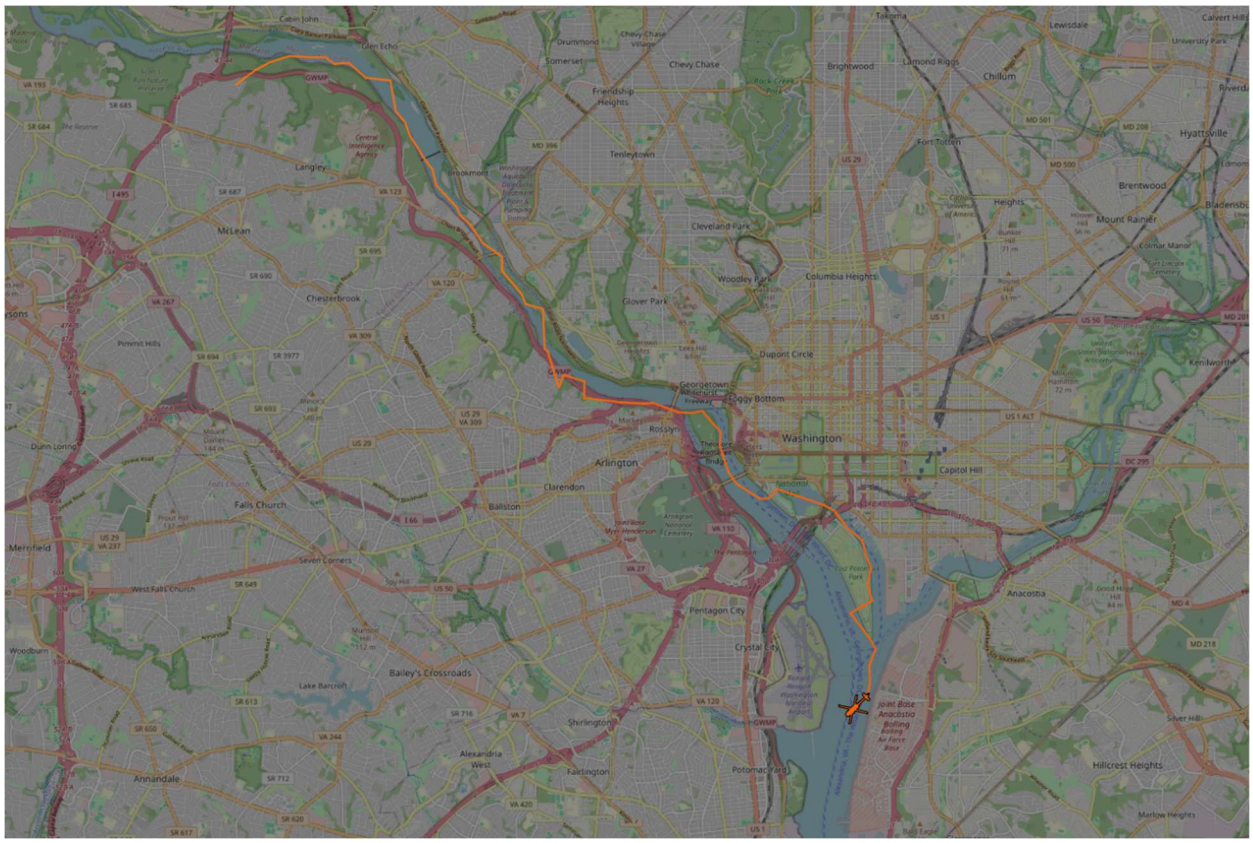

Figure 32. Mode S MLAT data (orange-colored trace) for aircraft address AE313D on the night of the accident flight. (Source: ADS-B Exchange)

A review of historical ADS-B data for the accident helicopter, which was gathered by the FAA on February 19, 2025, showed that there was no historical ADS-B data for the past 730 days.[42] Based on MLAT data populated in publicly available ADS-B databases, **Table 8** shows the aircraft address behavior for the accident helicopter's transponder from December 14, 2022 until January 29, 2025.

Table 8. The accident helicopter's aircraft address behavior from December 14, 2022 to January 29, 2025.

Date(s)	*Aircraft Address*
December 14, 2022 to October 5, 2023	AE213D
October 19 and 20, 2023	AE313D
December 7 to 14, 2023	AE213D
December 18, 2023 to January 4, 2024	AE313D
February 22, 2024	AE213D
November 13 and 14, 2024	AE313D
November 14, 2024	AE213D
January 29, 2025	AE313D

9.2 ADS-B Out Anomaly with 12th BN Helicopters

ADS-B historical data provided to the NTSB by the FAA for the 25 helicopters assigned to the 12th BN showed that 8 helicopters, including the accident helicopter, did not have a recent history of radiating ADS-B.[43] All 8 helicopters had their APX-123A transponder installed in the April to May 2023 timeframe.[44] The accident helicopter had no ADS-B historical data since its APX-123A transponder installation. The remaining 7 of the 8 helicopters showed ADS-B historical data shortly after their APX-123A transponder was installed, but their ADS-B historical data stopped at various times between May 2023 to December 2023. All 8 helicopters that exhibited this behavior were UH-60L models. All of the UH-60M helicopters at the 12th BN had recent ADS-B activity.

9.3 UH-60L Tail No. 92-26446 Evaluation

On March 5, 2025, the Helicopter Airworthiness Group Chairman, an electrical engineer from the NTSB Recorders Lab, and a representative from the FAA met with the 12th BN Aviation Maintenance Officer at DAA in Fort Belvoir, VA to examine the transponder installed on UH-60L tail no. 92-26446. Tail no. 92-26446 had its APX-123A transponder installed on April 17, 2023 and its last ADS-B transmission recorded by ADS-B receivers was on December 4, 2023.

[42] The FAA checked both aircraft addresses AE213D and AE313D.
[43] The FAA gathered this ADS-B data on February 19, 2025. The timeframe of the ADS-B historical data is 730 days from the date the data is obtained.
[44] According to the 12th BN, the APX-123A transponders were installed on the battalion's helicopters by a contracted organization.

The helicopter was powered up using the APU, with no noted faults on the RCU at power up. Mode S showed as off after power up; this was the last configured state for Mode S from the last flight. Modes 3/A and C showed as active after power up. The group turned on Mode S via the MS button on the RCU. Navigating to the ADS-B squitter setting on the RCU, the ADS-B squitter setting was "ON". The aircraft address on the RCU showed its correctly assigned aircraft address. The "power down" setting showed as "SAVE". A built-in test (BIT) was manually activated and the RCU displayed no faults.

Subsequently, APU power was removed. The RCU was partially removed from its installed location and pins 12 through 35 on the cannon plug for the J1 connector cable to the RCU were checked to verify its aircraft address configuration, with pin 34 used as a reference for known ground (logic "0"). The binary aircraft address read from the cannon plug matched that of the helicopter's assigned aircraft address. A second check of pins 12 through 35 resulted in identical results to the first check.

9.4 ADS-B Troubleshooting by 12th BN

A collaborative effort between NTSB, FAA, BAE Systems, and the 12th BN was accomplished in March and April 2025 to isolate the cause of why the 7 UH-60L helicopters were not broadcasting ADS-B data. Examination of these 7 helicopters by the 12th BN found their ADS-B squitter setting on the RCU was set to "ON". Additional troubleshooting found that their time source setting on the RCU was set to "153" and not the required setting of "059". Correcting the time source setting on the RCU to "059" resulted in successful transmission of ADS-B data to ground stations during subsequent flights when Mode S was active. At the time of this report, the ADS-B broadcast discrepancy had been resolved on 6 of the 7 helicopters at the 12th BN, with the 7th helicopter still undergoing heavy maintenance.

9.5 MWOs for UH-60L Transponders

MWO 1-1520-237-50-105 for the installation of the APX-123 transponder contained instructions for a MOC to be accomplished after installation of the transponder. The MOC instructions included steps to set the time source on the RCU to the "153" setting.

MWO 1-1520-237-50-118 for the installation of the APX-123A transponder also contained instructions for a MOC to be accomplished after installation of the transponder. The MOC instructions included steps to set the time source on the RCU to the "059" setting. Additionally, the MOC instructions included steps to set the ADS-B squitter on the RCU to "ON". Additionally, the MOC instructions included steps to use a transponder test set to confirm proper functionality of ADS-B, specifically that

the test set received the flight ID, aircraft address, and the GPS position and altitude information.

9.6 Corrective Actions

Based on the findings of this investigation, on April 25, 2025, US Army Aviation and Missile Command (AMCOM) released Aviation Safety Action Messages (ASAMs) for both fixed and rotary wing aircraft that directed the accomplishment of operational checks of the aircraft transponder, to include ADS-B out functionality (if the aircraft is equipped with ADS-B) and verification of the assigned aircraft address.

E LIST OF ATTACHMENTS

Attachment 1 – NTSB Materials Lab Email Report for examination of the loose fitting on a pitot outlet line

Attachment 2 – NTSB Materials Lab Email Report for the evaluation of the dial faces of the barometric and radar altimeters

Attachment 3 – NTSB Materials Lab Email Report for the examination of the J1 connector of the RCU

Attachment 4 – Tail no. 00-26860 aircraft logbook excerpt for the last PMS, PMI-1, and PMI-2 inspections

Attachment 5 – Tail no. 00-26860 aircraft logbook excerpt for the last 12-month inspection of the pitot-static system

Attachment 6 - Tail no. 00-26860 aircraft logbook excerpt for the pitot-static system discrepancy dated September 13, 2024

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